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FLY-BY-WIRE TECHNIQUES

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SPERRY PHOENIX COMPANY

TECHNICAL REPORT No. AFFDL-TR-67-53

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FOREWORD

This report was prepared by the Sperry Phoenix Company, Phoenix, Arizona on Air Force Contract AF33(615)-3615 under Task No. 822510, "Unique and Promising Flight Control Actuation Techniques," of Project No. 8225, "Study and Research on Fly-By-Wire Techniques". The work was administered under the direction of the Control Elements Branch of the Flight Control Division, Air Force Flight Dynamics Laboratory, Research and Technology Division, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, by Mr. V. R. Schmitt and F/L J. P. Sutherland, Project Engineers. Work on this contract was performed between 15 March 1966 and 15 February 1967. The manuscript was released by the authors in June 1967 for publication as a technical report.

Mr. F. L. Miller was the principal investigator. Assistance was received from Mr. J. E. Harrison on flight controls and from Messrs. J. E. Emfinger and D. C. Cook on the simulations. The work was performed by members of the Research Department of the Sperry Phoenix Company. Special thanks are extended to all those in the aircraft industry who consulted with us on mechanical and fly-by-wire control systems.

This report concludes the work on the contract. The Sperry Phoenix report number is LJ-1201-0723 and has been reviewed and approved by Sperry Phoenix Company.

This technical report has been reviewed and is approved.

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ABSTRACT

Manual flight control systems are described in which the sole means of control between the pilot's station and the control actuator is in the form of electrical signals. No mechanical control links are used in the system. Such a system where vehicle motion is the controlled parameter is defined as a fly-by-wire control system. Because of the growing number and severity of problems in mechanical control systems, particularly in large and high speed aircraft, fly-by-wire systems are evolving out of necessity. Fly-by-wire control is shown to provide many advantages over conventional mechanical flight control systems. Principally, they are reduced weight and volume, improved control performance, reduced design effort and maintenance time, the feasibility of standardizing flight control systems, and reduced vulnerability. System design requirements and tradeoffs are discussed such as the types of components used, control signal format, method of transmitting signals, actuator configurations, degrees of redundancy, failure detection techniques, and artificial feel mechanization. Examples are given of the application of fly-by-wire control to various classes of aircraft. The primary benefits derived depend on the class of aircraft. Control system technology has reached the point where practical fly-by-wire system designs can be realized today. The next logical step in its development is to build and fly a fly-by-wire system to demonstrate its feasibility and after many flight hours to provide in-flight proof of its maturity.

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LIST OF SYMBOLS AND ABBREVIATIONS

SYMBOLS

A	Actuator piston area
a_0, a_1, b_0, b_1	Constants to describe airframe characteristics
$C*$	Blend of pitch attitude rate, pitch attitude acceleration, and normal acceleration
$\bar{C}$	Mean wing chord
C_{m_α}	Pitching moment coefficient due to angle of attack
$C_{m\delta_e}$	Pitching moment coefficient due to elevator
C_{z_α}	Normal force coefficient due to angle of attack
$C_{z\delta_e}$	Normal force coefficient due to elevator
F	Force
F_s	Control stick force
G	Transfer function or variable gain
g	Gravity
I	Servo amplifier current
I_{yy}	Pitch axis moment of inertia
$K_{n_z}, K_{\dot{\theta}}$	Constants to describe airframe characteristics
M	Mass, Mach number
M_{δ_e}	Pitching moment due to elevator
M_q	Pitching moment due to pitch rate
$M_{\dot{w}}$	Pitching moment due to normal acceleration
M_w	Pitching moment due to normal velocity

n	Load factor
M_o	Aircraft neutral point
n_z	Aircraft normal acceleration
p	Pressure
Q	Flow rate
q	Dynamic pressure, pitch attitude rate
S	Wing surface area
s	Laplace operator
t	Time
u_o, v_o	Aircraft steady-state longitudinal velocity
v	Change in aircraft velocity
X_{CG}	Aircraft center of gravity in percent of mean chord
Z_ω	Aircraft normal force due to normal velocity
Z_{δ_e}	Aircraft normal force due to elevator
α	Angle of attack
Δ_s	Change in stabilizer position
δ_A	Intermediate actuator displacement
δ_e	Elevator deflection
δ_s	Control stick displacement
ζ	Damping ratio
$\dot{\theta}$	Pitch attitude rate
ρ	Air density
ω	Angular velocity, constant used in aircraft description

ABBREVIATIONS

AAFSS	Advanced aerial fire support system
AFCL	Advanced flight control linkage
AFCS	Automatic flight control system
AMSA	Advanced manned strategic aircraft
AWL	All-weather landing
BITE	Built-in test equipment
CAS	Command augmentation system or control augmentation system
EFCS	Electrical flight control system
FBW	Fly-by-wire
IHAS	Integrated helicopter avionics system
LEM	Lunar excursion module
LRU	Line replaceable unit
LVDT	Linear variable differential transformer
MTBF	Mean time between failures
PIO	Pilot induced oscillations
RIG	Rate integrating gyro
SAS	Stability augmentation system
SST	Supersonic transport
VTOL	Vertical takeoff and landing aircraft

SUBSCRIPTS

ϵ	Error signal
M	Signal has been shaped

SECTION I

INTRODUCTION

Aircraft design is about to enter a new era in which the mechanization of flight control systems will be electrical rather than mechanical. Electrical flight control systems are commonly known as fly-by-wire systems. They will be integrated with the automatic flight control systems to provide better performing and more efficient military or civil aircraft. This report is an introduction to fly-by-wire control system design. It establishes the system requirements and design criteria for fly-by-wire control. It also establishes the types of components available today and the combinations that are best suited for mechanizing such systems.

A fly-by-wire flight control system is an electrical primary flight control system employing feedback such that vehicle motion is the controlled parameter. No mechanical backup is used. Fly-by-wire has been studied and proposed for at least the past 15 years, often under the title "Electrical Flight Control Systems". However, past research has nearly always been narrowly aimed at one or two specific approaches which replaced the link between the control stick and the surface and ignored the handling quality or feel requirements. This report satisfies a need for a more general approach to the subject.

Although mechanical control system designs have improved tremendously through the years both in techniques and materials, they have been having a progressively more difficult time in keeping up with the performance gains and control requirements of successive generations of aircraft. Most designers have agreed that fly-by-wire could solve the flight control problems if a practical approach could only be mechanized. The problem has been that no one has satisfactorily provided a practicable and reliable fly-by-wire system design that could be produced with existing hardware. This problem has several facets. One primary factor has been the unavailability of components having proven reliability. Another factor is that fly-by-wire design is a multidiscipline venture that encompasses mechanical, electrical, and hydraulic engineering. Further, the application of redundancy has generally not been well understood. This report will attempt to eliminate these factors to show how a practicable, redundant fly-by-wire system can be mechanized using available hardware.

This introduction is Section I of this report; Section II presents the sources of data included. Section III provides a historical perspective of the evolution of flight control systems including the previous and related work on fly-by-wire and problems involved. Many discussions were held with pilots and engineers in the aircraft industry to uncover problem areas in flight control systems and to determine their attitudes and past and planned work on fly-by-wire. Section IV discusses fly-by-wire control in general and several existing systems in particular, such as the X-20 system, and several pseudo fly-by-wire systems (having mechanical reversion) such as the F-111. A component discussion follows in Section V which describes available and preferred devices. Section VI describes the system design criteria and tradeoffs. Candidate systems are described which satisfy the design criteria. The results of simulation studies of these systems are described in

Section VII, including limited breadboard model work. In Section VIII, the mechanical control systems and equivalent fly-by-wire systems for the B-52, F-111, and CH-46 are compared to show the relative benefits for several different classes of aircraft. After the conclusions and recommendations, which are found in Section IX, a glossary of terms is presented in Section X that establishes a much needed common vocabulary.

SECTION II

SOURCES OF DATA

1. LITERATURE

Because of the relative newness of the art of fly-by-wire design, no single preferred source of data is available. Therefore, all likely information sources were investigated during the program period to uncover data on past, present, and planned projects. To cover as many sources of related work as possible, both a literature search and plant visits were carried out. The literature search included an Abstract Bibliography Request from the Defense Documentation Center and hand searches of the Technical Abstract Bulletin, Science and Technical Aerospace Reports, International Aerospace Abstracts, The Engineering Index, and the Applied Science and Technology Index. The results of the search are found in References and Bibliography, Section XI, at the end of the report.

2. PLANT VISITS

Because the greater share of work relating to fly-by-wire design has never found its way into the literature, plant visits were made to various airframe and actuator companies to determine their attitudes, past work, or plans (if any) involving fly-by-wire control. The following companies were visited and the personnel contacted are listed as follows:

29 April 1966 Grumman Aircraft Company, Bethpage, New York

Mechanical Systems Section
- J. Leonard
- J. Morgan
- T. Cosbey
- R. Magner
- A. Sammis
- H. Shephard

26 July 1966 The Boeing Company, Seattle, Washington

Control Dynamics Group
- D. Bird
- R. Hare
- R. Hurlow
- D. Lewis
- H. Toby

28 July 1966 North American Aviation, Los Angeles, California

B-70 Division
- J. Campbell
- B. Palarz
- C. Crother

28 July 1966 Douglas Aircraft Company, Long Beach, California

Flight Controls Group
V. Sethre
G. Schlanert

22 November 1966 General Dynamics, Fort Worth, Texas

F-111 Control System Group
H. Z. Scott

30 September 1966 Hydraulic Research and Manufacturing Company, Burbank, California

J. Stuart
D. Wood
G. Jenny

23 June 1966 National Water Lift Company, Kalamazoo, Michigan

Research and Development Department
C. Hawk
R. Salemka
V. Heine

1 November 1966 Weston Hydraulics Limited, Van Nuys, California

D. Irwin

3. RELIABILITY: THE NUMBERS GAME

A word of caution on reliability data is in order at this point. Anyone who has worked with reliability enough to be familiar with the various data sources soon realizes that reliability may degenerate to a numbers game because of unreliability of the data. This is particularly true for nonelectronic component data. The confidence level of electronic component data can be made tolerably high because enough life test time can be accumulated on very large numbers of parts to constitute a significant sample. A readily available source for such data, MIL-HDBK-217A (Ref 1), provides an acknowledged common reference to back up arguments. Unfortunately this is not true for nonelectronic components. Several widely quoted references (Ref 2, 3) exist based on field data which have been accumulated in an attempt to bring some order out of chaos in this area. Because of the lack of better (or any) data, these sources are too often quoted incorrectly for the sake of quoting a source, with the hope that it will add some credence to the argument at hand. For example, an entry in reference 2 on component failure rates lists: "mechanical assembly $\lambda = 18.3 \times 10^{-6}$". This value came from the linkages and mechanisms of a bombsight; that is, a mechanical computing mechanism. However, this number has been applied to many different types of linkages ranging from a single actuator mechanical feedback link to an entire flight control system. A second problem exists in the wide range of values which can be found for a particular component, for example, an electrohydraulic servovalve. The value ranges from 5×10^{-6} (Cadillac Gage) to 1.5×10^{-3} (Ref 3). Nothing is said in the sources about the type of valve, its

application, environment, or size. Further, the data are for valves in service years before the source publication, and the valves were designed several years before that. Hence, the data is anything but up to date. The failure rate quoted in Avco for a servo amplifier of 37×10^{-5} is another misused figure since it refers to vacuum tube amplifiers which are hard to find these days except in the older aircraft. An up-to-date value for a transistor amplifier ranges from 4 to 12×10^{-6} each hour depending on the design and application.

The point of this discussion is that data cannot be meaningfully quoted from a source unless the source's application, environment, component type, etc are known, and the quoter's application, environment, component type, etc are similar. For this reason, the values employed in this report are not quoted directly from any one source except for the electronics. Rather, a reasonable estimate has been made based on all known source data plus unpublished manufacturers' data where it is available. If the reader has access to better data with an acceptable confidence level, he should make appropriate corrections.

SECTION III

HISTORICAL PERSPECTIVE AND PROBLEM DISCUSSION

1. FLIGHT CONTROL SYSTEM DEVELOPMENT

a. Control Linkages

In early days flying was solo in small, slow airplanes. Flying has developed over the past 60 years until today aircraft carry upwards of 200 people at speeds up to 5 to 10 times as fast as ground transportation. Designers have done an excellent job of providing the pilot with controls that can handle such craft when necessary, with powered assistance, i.e., mechanical reversion. The success of mechanical controls has prompted many people in the field to believe that all future craft should be so controlled. Our belief is that this attitude can compromise controls design such that flight safety will be reduced and system complexity will be increased unduly.

Ever since the Wright brothers invented hinged control surfaces, people have been looking for ways to reduce the moment required to move them. Such developments include the adjustable stabilizer and aerodynamic balances. Fixed balances (such as offset hinges, horns, overhangs, etc) soon gave way to adjustable devices because the pilot could no longer cope with the incremental forces on the larger and faster aircraft. Movable surfaces (i.e., tabs) solved this problem for a good while, but they in turn have reached their limit in effectiveness.

Powered controls came about originally as part of the autopilot. A limited authority actuator would move the controls for the pilot to maintain level flight thereby lightening his work loads. With progressive increases in aircraft size and speed, power boost became necessary to fully utilize the available maneuverability. Fully powered controls came into being shortly after World War II. Such controls are completely irreversible since the pilot is no longer directly connected to the surfaces.

When power assist was added to the controls, control reversibility was reduced and so was the pilot's control feel. Therefore, feel was augmented artificially with springs and dashpots. When the controls became fully powered, all feel was lost. Therefore, all of the pilot's cues had to be supplied artificially. While it is true that artificial feel is not required in moving the control surfaces, it is required to give the pilot the proper handling quality characteristics for control of the aircraft. The artificial feel system then becomes an important and integral part of the flight control system. The stability augmentation system and autopilot must also be included in these considerations since they alter the basic dynamic and static stability and hence the handling qualities and feel characteristics of the aircraft. This subject must be understood before adequate approaches can be formulated.

The evolving control system has been further complicated by the addition of stability augmenters and control stick steering. The latter was added to reduce the effects of friction, inertia, and nonlinearities in the

control system. The simplified control system shown in figure 1 depicts the simple reversible system which is still used in light aircraft today. Here all control forces are reflected back to the pilot's hand. This is no longer the case in the irreversible fully-powered system shown in figure 2. Therefore, an artificial force producer must be added as previously mentioned. Control of the neutral position of the feel system as it changes with flight condition is also required. This is a trim function very similar to the simple system. A parallel input servo, figure 3, moves the control stick along with the pilot. Such a servo commonly provides AFCS (Automatic Flight Control System) inputs so that the pilot can observe and monitor its actions. The series input servo, figure 4, adds to or subtracts from the pilot's inputs so that no control stick motion occurs. This type of servo is commonly employed for stability augmentation signals which would otherwise cause considerable high frequency activity at the control stick. Such motion would be very annoying to the pilot. Figure 5 shows another technique for adding series inputs which results in a lighter control system. Figures 6 and 7 show two types of control stick steering mechanizations. These schemes are also called command or control augmentation systems. Their purpose is to improve control response by bypassing control system friction, inertia, deadzones, etc, and any other troublesome problem that the particular control system might present. Such systems have been in use since about 1960 when designers finally decided that perhaps the mechanical control system could no longer cope with the control requirements of the aircraft. Control stick steering or command augmentation, which is used on the F-111, the A-7A, the supersonic transport (SST), and the jumbo jets (i.e., the C-5A and its derivatives) among others, is the forerunner of the fly-by-wire control system shown in figure 8. The control stick steering system has now been refined to the point where it is in essence a fly-by-wire control system with mechanical reversion. The last step in this evolution is to remove the backup system.

b. Mechanical Control Systems Problems

The relatively simple direct linkages, cables, and feel springs for manual control described cannot meet the greater demands of advanced aircraft control system design requirements. Simple manual control systems have been replaced by complex nonlinear linkages, mixing assemblies, power actuation devices, and active artificial feel systems. These complex manual control systems have increased requirements for space and weight in aircraft where both are at a premium. Nonlinearities such as deadband, hysteresis, and backlash result from the increased compliance, inertia, and friction of complex mechanical devices. These nonlinearities degrade the performance of the control system, and as a result, the full capabilities of the aircraft are not realized. Additional control problems also result from temperature variations and airframe flexibility. Now that full-time, full-power stability or command augmentation systems (SAS or CAS) have arrived, the mechanical system is used only in the event of a CAS failure. In certain aircraft, the mechanical system may not even provide survival capability because of inherent aircraft instabilities at some flight conditions. For safety and mission reliability, therefore, the augmentation system must be as reliable as the mechanical control system. Thus, the mechanical control system imposes a weight and space penalty at certain flight conditions where it provides no usable function. The solution to these problems is to replace the mechanical control system with a fly-by-wire system.

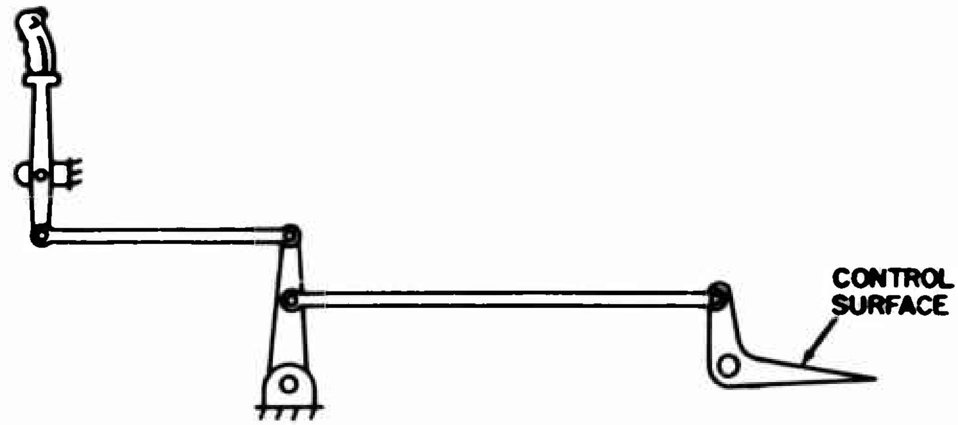


Figure 1
Simplified Aircraft Control System

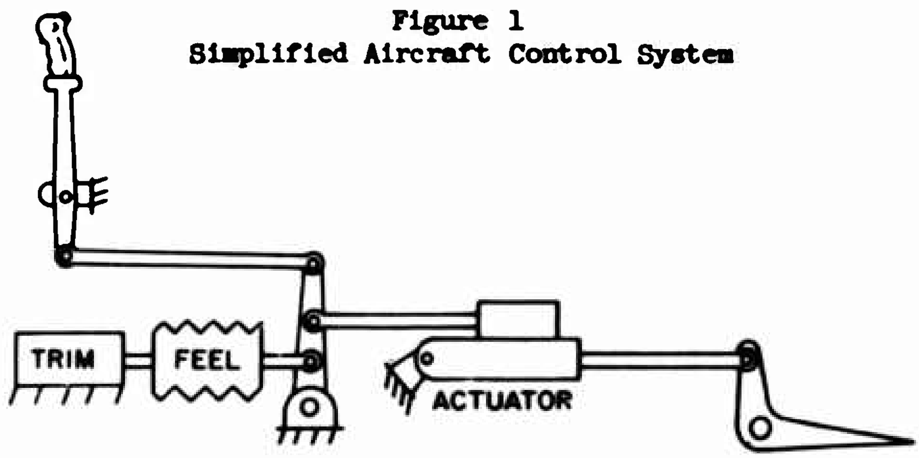


Figure 2
Fully Powered (Irreversible) Control System

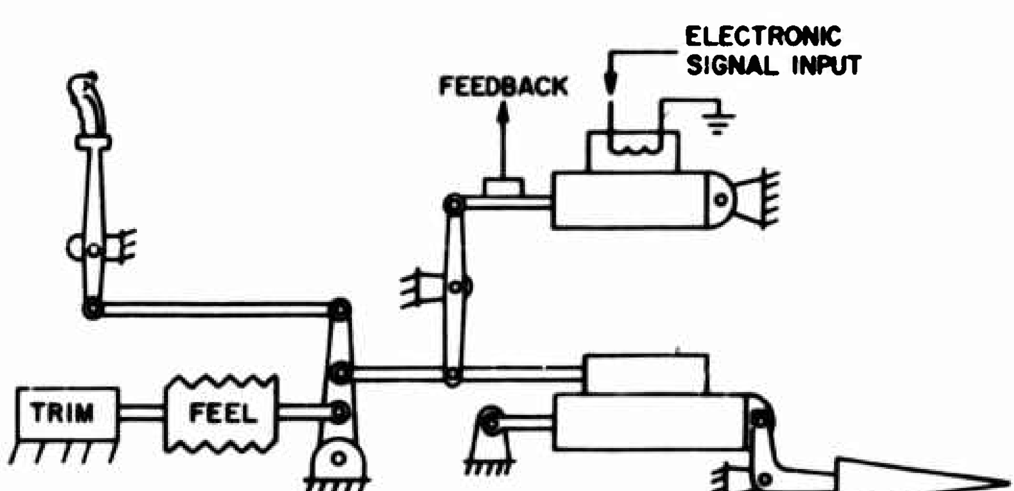


Figure 3
Parallel Input Servo

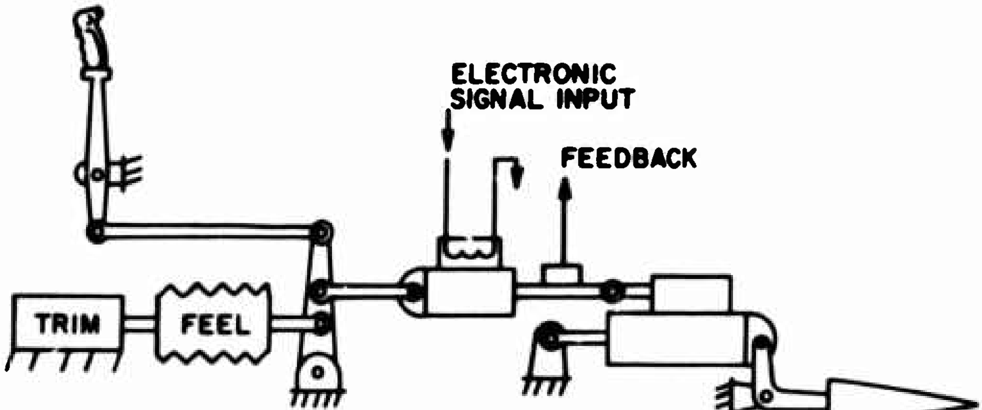


Figure 4
Series Input Servo

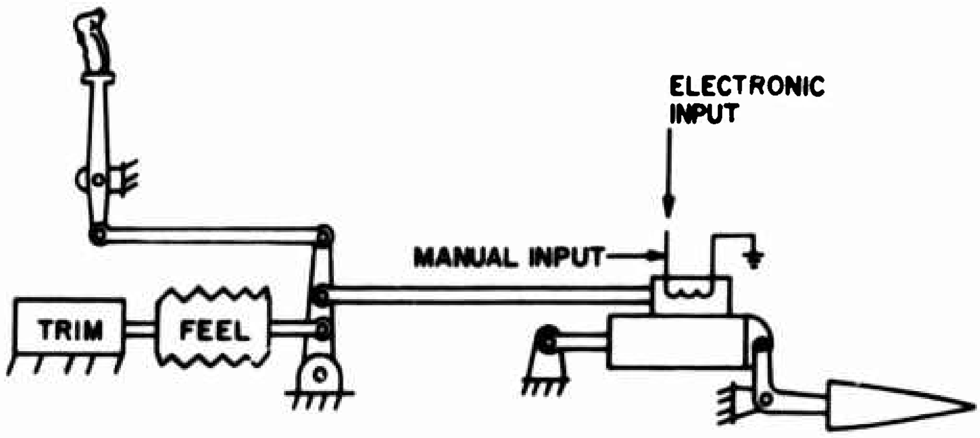


Figure 5
Dual Valve Input

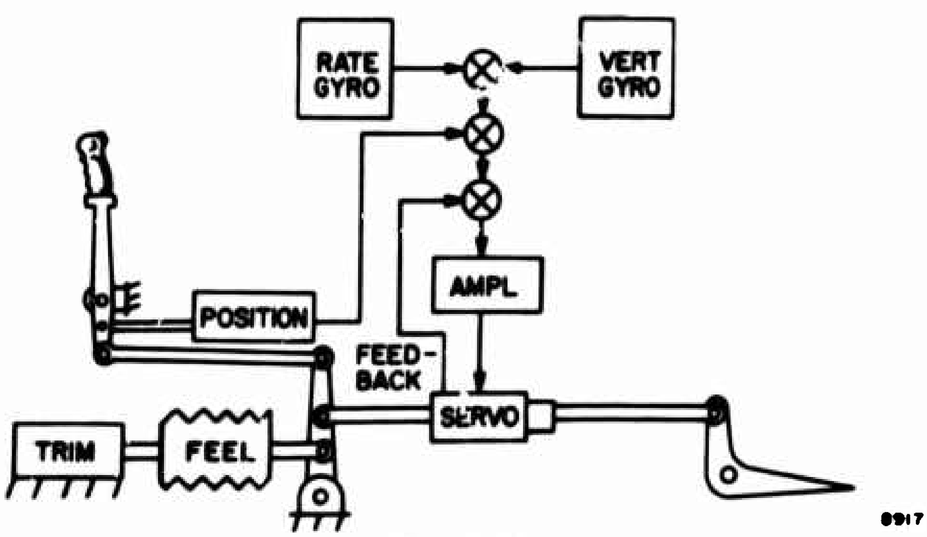


Figure 6
Differential Servo Type of Control Stick Steering

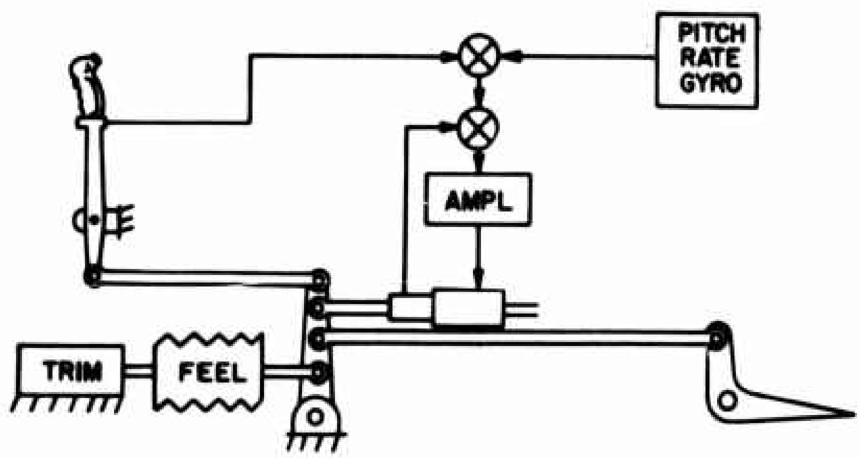


Figure 7
Force Stick Type of Control Stick

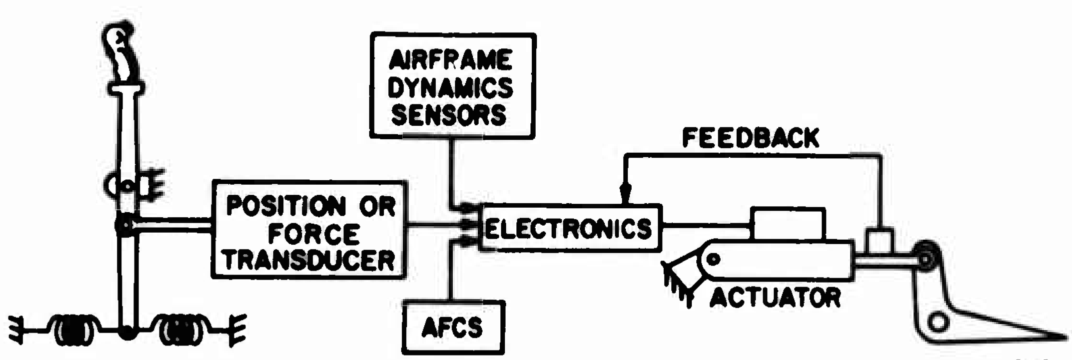


Figure 8
Fly-By-Wire Control System

The complexity of modern control systems is readily apparent in figure 9 which presents a slightly simplified diagram of the F-111 mechanical pitch and roll flight control system. This system is a backup to a primary electrical flight control system which is itself a fly-by-wire system. General Dynamics recognized the potential difficulties of the mechanical system when they relegated it to a standby role. The following discussion will set forth the most notable problem areas that occur in mechanical systems along with some case histories. The application of fly-by-wire to eliminate these problems is obvious in most cases.

The controls designer must consider a large number of characteristics and factors in establishing a flight control system design. Some of these characteristics are required by the system. These typically include:

(1)	Nominal travel	(6)	Minimum increment of control
(2)	Operating loads	(7)	Positioning accuracy
(3)	Maximum velocity	(8)	Synchronization accuracy
(4)	Frequency response	(9)	Stability
(5)	Sensitivity	(10)	Life

A number of negative characteristics also exist that tend to prevent the designer from attaining his design goals. These typically include:

(1)	Friction	(7)	Inertia
(2)	Temperature change	(8)	Compliance
(3)	Deadbands	(9)	Body bending
(4)	Hysteresis	(10)	Routing
(5)	Backlash	(11)	Weight
(6)	Complexity	(12)	Volume

The order of importance depends on the particular aircraft and does not necessarily follow the above order. The significance of most of these problem factors grows along with the size of the airplane. For example, small fixed-wing fighter/attack aircraft such as the F-5 and A-6A have relatively few problems. Such aircraft in general would not benefit appreciably from the application of fly-by-wire except to reduce control inertia, weight, and volume. On the other hand, although all present VTOL designs are relatively small, they would benefit appreciably from fly-by-wire because of the reduction in weight and complexity. Fly-by-wire would typically reduce VTOL weight by several hundred pounds and in some cases provide control designs that are virtually impossible to obtain mechanically.

Friction is one of the biggest bugaboos of the control designer. Low system friction is essential to ensure that an otherwise excellent airplane is not made unacceptable from a handling qualities point of view. Excessive friction masks the control feel characteristics; this is particularly critical around neutral or trim position. Excessive control breakout forces due to friction is particularly damaging to proper control feel. To provide positive centering, a preload force larger than the friction force is required. Preloads over 1-1/2 pounds are excessive since they produce a notch effect and make simple tracking tasks very difficult. In very large aircraft such as the C-5A, the breakout friction can be so high that it exceeds the average pilot's capability to even move the controls. The SST will be nearly as bad.

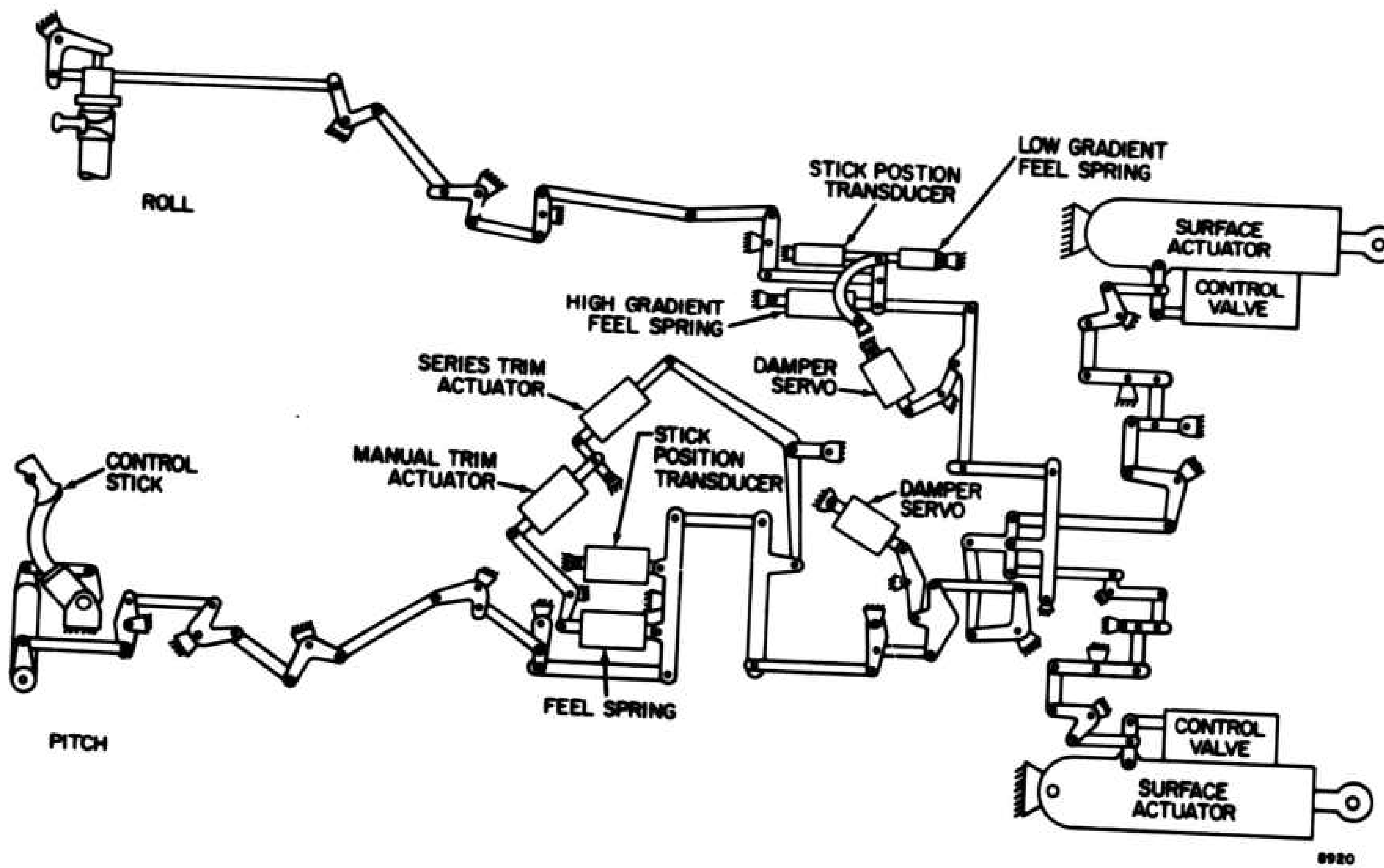


Figure 9
F-111 Pitch/Roll Mechanical Control System

Estimates of performance on the mechanical system alone show that ±1 g acceleration overshoots are extremely likely. The common solution to these problems is to add a small parallel boost actuator to help the pilot overcome the friction. Stick force transducers are now also required to control the boost actuator. The actuator force output is limited to just below the friction level so that a small additional force applied by the pilot moves the controls. Friction causes control system hysteresis which significantly affects how closely the system returns to the trim position once it has been displaced. If the pilot trims his stick forces for a particular trim speed, the hysteresis could allow the speed to change significantly before an effect is felt at the stick.

Any moving component in the control system adds friction. This includes bearings, cable systems, hydraulic servovalves, cockpit pressure seals, cartridge preload springs, fairleads and antibacklash springs. By actual count, the F-111 system shown in figure 9 contains 114 bearings, 2 servovalves, 2 cockpit pressure seals and 3 feel springs. The factor that saves this system is the unique bearing used that is self-lubricating and has an extremely low friction level. Cockpit pressure seals presented a large problem in at least one transport aircraft when over a period of time tobacco tars accumulated in the seal lubricant and caused a dangerously high friction level. Cable system friction is usually higher than pushrod system friction. It depends on the number of pulleys and quadrants, travel, and the rigging load. Tension regulators are required to maintain the desired tension load to minimize friction.

Temperature changes also cause friction because of unequal expansion between the airframe, which is aluminum, and the control system, which is basically steel. In aircraft traveling below Mach 2, the temperature change is due to altitude. The temperature differential ranges from over +120°F (+48.9°C) on the ground to -85°F (-65°C) at high altitudes. The Martin Seamaster, for example, could not fly higher than 25,000 feet because unequal expansion would lock the control system. At one time the Convair 880 had a dual cable consisting of an aluminum tubing over a spring. This system would also lock up at high altitudes. Aircraft flying at Mach 3 or higher, such as the B-70, A-11, and SST have a different temperature problem because of aerodynamic heating. At Mach 3 the skin temperature of the aircraft rises to over 500°F (260°C). Typically, the fuselage of the SST and B-70 grows 12 inches at sustained Mach 3 speeds.

Temperature and moisture combine to cause icing which can affect the q-springs used in artificial feel systems. Moisture enters the pressure vents and freezes on the bellows to lock the spring. The effects vary with the system ranging from locked controls to soft, spongy controls.

Backlash and deadbands are very similar; backlash is free play and deadbands are thresholds such as preload. Backlash is the more common problem because it normally results from wear at bearings and joints. Backlash in the B-47 reportedly has been as large as 1/2 inch after a year of flying. Backlash effects range from sloppy and unsatisfactory control characteristics to PIO (pilot induced oscillations).

Complexity means an increase in the number of individual components in the control system. This increases cost, weight, volume, failure rate, spare parts, and maintenance. Control system complexity is primarily caused

by the greater performance requirements of modern flight control systems. Another cause of complexity is the very inefficient design practice of stuffing the control system into whatever space is left after all of the other subsystems are in the airplane. As the C-5A and SST designers have no doubt discovered, control system redundancy is yet another cause that has significant cost, weight, and space penalties for a mechanical system. Until now the mechanical systems have been low performance and easy to maintain, but this is no longer true in the newer aircraft.

Control complexity has a very decided effect on weight and design tractability in VTOL aircraft as mentioned earlier in this discussion. The thrust vector hover control for lift fan systems is a particularly complex and difficult mechanical design problem. The mixing mechanism in tilt-wing or engine designs used for transition from hover to cruise is large, heavy, and also complex. These two areas are prime candidates for early application of fly-by-wire techniques. Weight savings are particularly important in VTOL aircraft because it improves the all-important power-to-weight ratio.

The complexity of the swing wing used on the F-111 and the Boeing SST causes a particularly difficult design problem for the lateral control system. The variable wing sweep mechanization requires a variable attachment point for the spoiler control system linkage which consequently becomes a messy design problem. General Dynamics chose to solve this problem on the F-111 by employing fly-by-wire techniques. No system failures or unusual problems have occurred as of this date with over 2,000 flight hours being logged.

Control cable routing in the B-70 presented some difficult problems because of the long runs, limited space, and friction effects. The throttle system could not operate with the deadbands, hysteresis, and backlash inherent in a mechanical cable system. The acceptable solution was to use fly-by-wire control. The flight control system was not allowed to go fly-by-wire (which seems inconsistent) although the cable routing problems were considerable. The presence of the fuel cells in the fuselage complicated the routing problem. All available routes either involved going outside of the airframe or using an excessively large number of direction changes which would result in intolerable friction and backlash. The solution was to use straight runs down the fuselage through the fuel cells. Special seals were provided to allow this path to be used. The expense and problems of going this route are obvious.

Another problem involving control routing and fuel cells occurred in the A-6A. The elevator and rudder control rods in this airplane are routed along the top of the fuselage between the aft fuel cell and the airframe. On occasion the fuel cell vent would stick while the aircraft was climbing. The trapped air would expand the cell enough to trap the control linkages against the airframe thus locking the controls. Usually the vent would open in a short time to restore normal control, but all pilots were not so lucky. Some had to leave their airplanes.

Control compliance and body bending (which is airframe compliance) cause similar problems, namely, deformation of the control system and/or the airframe when loads are applied. The effect is that of adding a spring inside the control loop which is destabilizing. The added phase lag commonly reduces system performance, but it sometimes causes PIO. A much more severe effect occurs when control system and airframe vibrational modes couple into each other. The modes then reinforce each other, causing overstress or fatigue in the airframe. This can either destroy the airplane or reduce its life considerably.

The solution to these problems is to replace the mechanical control system with fly-by-wire control. This presents a new set of problems in designing practical fly-by-wire systems within the present state of the art in controls design and components. The new problems are easier to solve, however, as is discussed in later sections of this report.

c. Artificial Feel

Artificial feel is a very important and integral part of the flight control system. In any control system design problem, the designer is given a relatively fixed plant (the aircraft in this case) for which he must design a controller (flight control system) that will cause the plant to behave in a desired manner for specified inputs (the desired flight maneuvers). The controller for a manned aircraft actually includes the pilot, but the controls designer cannot do much about the design of the pilot any more than he can do anything about the design of the aircraft (often to his chagrin). But he can do a great deal about the design of the flight control system which, since it primarily concerns artificial feel for the pilot, will be referred to as the artificial feel system.

An extensive treatment of artificial feel is beyond the scope of this report. Therefore, the following discussion will be of an introductory nature only. For a more detailed discussion, see references (4), (5), (6), (7), (8) and (9). To begin the discussion of artificial feel systems and its philosophies, it would be worthwhile to clarify the reason for using artificial feel. The trend toward powered controls classically has been considered to be the result of the high aerodynamic hinge moments associated with higher performance aircraft. This is not the principal reason for the use of powered controls. As the aircraft approaches the speed of sound, the aerodynamic characteristics of the vehicle change quite rapidly, and as the operational speeds move on into the supersonic region, the characteristics settle down to entirely different values than those for subsonic. This change is due to the aft movement of the center of pressure which at sonic velocities moves around in an irregular manner. The net result is that the stick forces are highly nonlinear and discontinuous in nature, both of which are unacceptable in terms of proper vehicle handling qualities. Accordingly, it has been in the best interest to divorce the stick forces entirely from the aerodynamic hinge moment forces, hence the birth of irreversible control systems and artificial feel. The use of irreversible control systems, therefore, is not because of the large hinge moments, but rather due to the deterioration of the stick forces and handling qualities near and above the sonic barrier. Artificial feel is then used to provide the proper handling characteristics to the pilot. With such a system, the pilot is completely isolated from the aerodynamic forces acting on the control surface. In the exact sense, a

fly-by-wire system is an irreversible system in which the method of transmitting the control signal is nonmechanical.

The artificial feel system has three purposes. First, it must provide the proper force and position cues to allow the pilot to obtain near optimum maneuver and path control. Second, it must aid in preventing inadvertent overstressing of the airframe. Third, the control motions under hands-off flight must result in satisfactory dynamic aircraft stability. The third requirement is satisfied by the stability augmentation subsystem. The first requirement is further subdivided into three areas: first, feel resulting from a change in airspeed from the original equilibrium speed which is given by the gradient of stick force per change in airspeed; second, feel resulting from normal acceleration during a steady-state maneuver which is given by the stick force required for a given acceleration; and third, the feel resulting from normal acceleration during a transient maneuver also given by the ratio of stick for per incremental acceleration.

Although the requirement for artificial feel has been firmly established, two separate philosophies exist on the manner in which characteristics should be obtained. One states that the artificial feel should duplicate the forces of the reversible control system since this is the framework around which all flying qualities have been tailored. In this scheme, the artificial feel varies the force gradient on the control stick as a function of flight condition, but it does not vary the aircraft characteristics. The other philosophy states that the forces of the reversible system need not be duplicated to provide the best handling qualities. In this method, the feel system fixes the control stick force gradient but varies the aircraft characteristics to change the response to commands as a function of flight conditions. Both methods use controllers to modify the pilot's inputs to the surface actuators so as to compensate for the aircraft's dynamic performance variation with flight condition. The first method generally uses open-loop control techniques. It measures the environment and/or flight condition (e.g., dynamic pressure or trim) and then uses these parameters to vary the control stick force gradient. The ratio of surface deflection to stick displacement remains constant. The success of this method depends on how well the variable force gradients match the desired gradients over the flight regime. For subsonic flight the open loop method can match the gradients satisfactorily. For transonic flight the match becomes much more difficult to obtain; consequently, the feel system generally requires a considerable degree of compensation and becomes complex.

The second method uses closed-loop control techniques since it feeds back the aircraft response (rate and acceleration) for summation with the command inputs. The controller consists of a fixed spring force gradient and model filter for shaping the command inputs. At this point the system sums the shaped commands with the aircraft rate and acceleration feedback to form what is basically an acceleration command system. Since specific surface positions are not commanded, the ratio of surface deflection to stick displacement varies with flight condition. Using the aircraft as an element in the forward path of a closed loop reduces the effects of its varying response characteristics. Using this basic advantage of a closed-loop servo, the system achieves a nearly constant acceleration response for a given stick displacement. In other words, it achieves a nearly constant stick force per

unit acceleration regardless of the flight condition. The gradient can be readily tailored electronically to suit the pilot by changing the model or the servo gradient.

The closed-loop method should be used for artificial feel in fly-by-wire systems for several reasons. First, it eliminates the reliance on air data computation which is notoriously unreliable. Further, it replaces heavy q-springs and/or bobweights with rate gyros and accelerometers which quite often already exist in the airplane for use by the AFCS or SAS. In addition, since the system provides feel characteristics that are nearly independent of the airframe, the feel of all aircraft within a class (e.g., fighters or bombers) can be standardized.

Factors in implementing an artificial feel system include friction, backlash, location of the column or stick and its displacement, whether series or parallel trim is used, airframe deflections, visibility, and the harmony between axes. They all have a bearing on how the system feels and performs. The complexity also depends on the particular axis. Because the pilot's feet are relatively insensitive to reaction forces, simple springs on the rudder pedals provide adequate feel for the yaw axis. Lateral axis control is much more sensitive so that spring forces are kept low and, consequently, so are friction and inertia. Again simple springs are usually adequate. The major problem occurs in the longitudinal axis. Therefore, the following discussion will be limited to that axis for a fixed-wing aircraft.

Basically, the control feel for a fixed-wing aircraft is measured by the response of the vehicle to a control command. The pilot can measure this response by a "feel" presented to him by the control stick, the acceleration forces on his body, and his visual attitude cues. The control stick feel will be provided by the force which he has to exert upon the stick and by the position of the stick relative to a straight and level trimmed position. In reviewing various authorities on pilot performance, they commonly agree that a pilot controls a conventional fixed-wing aircraft primarily by force. Position information also contributes, but it has a smaller influence.

For a given applied elevator stick force, the pilot would like the corresponding steady-state normal acceleration as illustrated by figures 10 and 11. Several basic requirements are illustrated by these figures. The pilot requires that the control stick have positive centering; this allows him to trim the vehicle to a reference flight orientation and stay in trim. It also reduces the possibility of inadvertent inputs and cross coupling. Once he has broken out of the positive centering area, the pilot would like a near linear relationship between incremental stick force and resulting normal acceleration. However, a means of limiting acceleration is required to prevent the pilot from over-controlling and causing structural damage to the vehicle.

The steady-state stick force gradient requirement for various operational altitudes is illustrated by figure 11. This constraint provides the pilot with a vehicle with constant response characteristics to a fixed force command, regardless of airspeed.

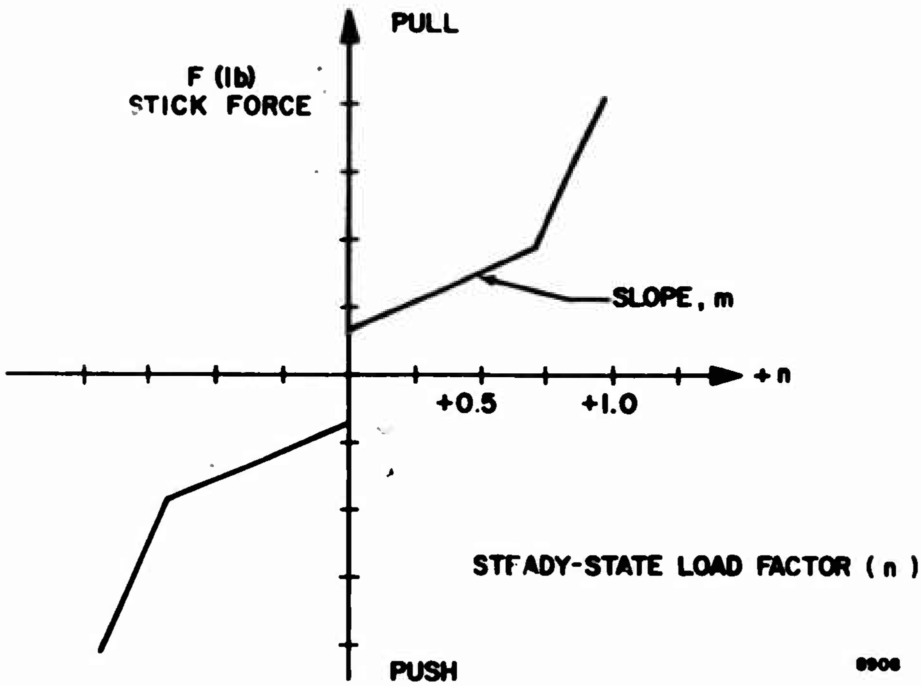


Figure 10
Elevator Stick Force Requirement

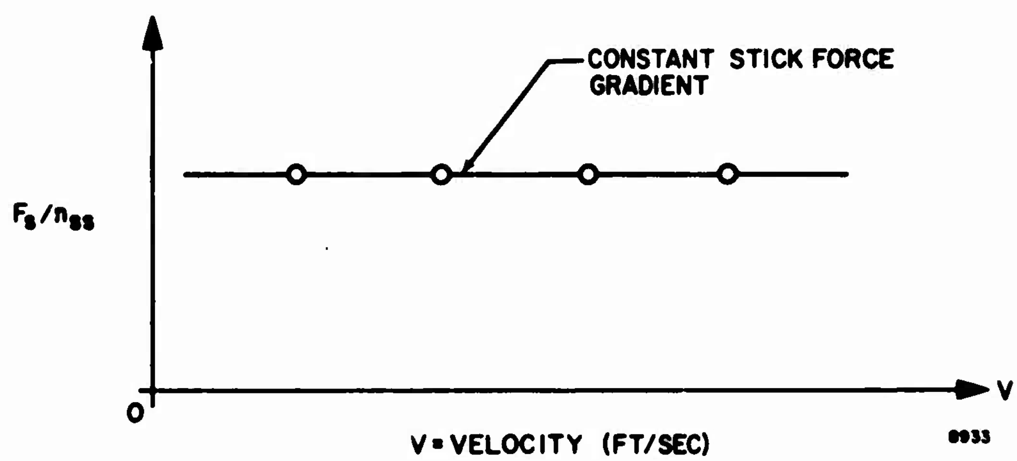


Figure 11
Constant Stick Force Gradient

The longitudinal stick forces and control feel, defined by figures 10 and 11 represent the requirements for a steady-state maneuver. In addition, for a given maneuver, the stick forces should be higher for a rapidly applied command than for a slowly applied one as shown in figure 12. This factor provides an additional feel indication of the acceleration magnitude of the commanded maneuver. If this characteristic were not provided, the structural limit load factor could be exceeded.

The following discussion looks at the factors that govern the selection of a more complex artificial feel system over a simple spring. As a simple model, consider the system as illustrated in figure 13. This system is an irreversible pitch control system for a conventional fixed-wing aircraft using a linear spring for artificial feel. The spring displacement, stick force, and elevator position can be approximated by the following relationships

$$F_s = +K\delta_s$$

$$\delta_s = K_1\delta e$$

$$F_s = +K_o\delta e$$

$$K_o = +K_1K \qquad (1)$$

where K represents the linear spring gradient; K_1 is the static gain between the stick position and the elevator surface.

Since the handling qualities of the pitch axis are generally referenced to the steady-state normal acceleration, we need the normal acceleration (steady-state) characteristic to a step elevator command. Of the various methods available for approximating the normal acceleration term, the simplest is to evaluate the steady-state response of the system transfer equation. Equation (2) presents the simplified transfer equation relating the normal acceleration characteristics to the elevator command.

$$\frac{n_z(s)}{\delta_e(s)} = \frac{-\left[M_{\dot\delta}Z_{\delta_e} + u_oZ_{\delta_e}M_{\dot w}\right]s + u_o\left[Z_{\delta_e}M_w - M_{\delta_e}Z_w\right]}{s^2 - \left[u_oM_{\dot w} + Z_w + M_q\right]s + M_qZ_w - u_oM_w} \qquad (2)$$

The steady-state response to a step elevator command may be defined by,

$$\left.\frac{n_z(s)}{\delta_e(s)}\right|_{s\to o} = \left[\frac{Z_{\delta_e}M_w - M_{\delta_e}Z_w}{M_qZ_w - u_oM_w}\right]u_o \qquad (3)$$

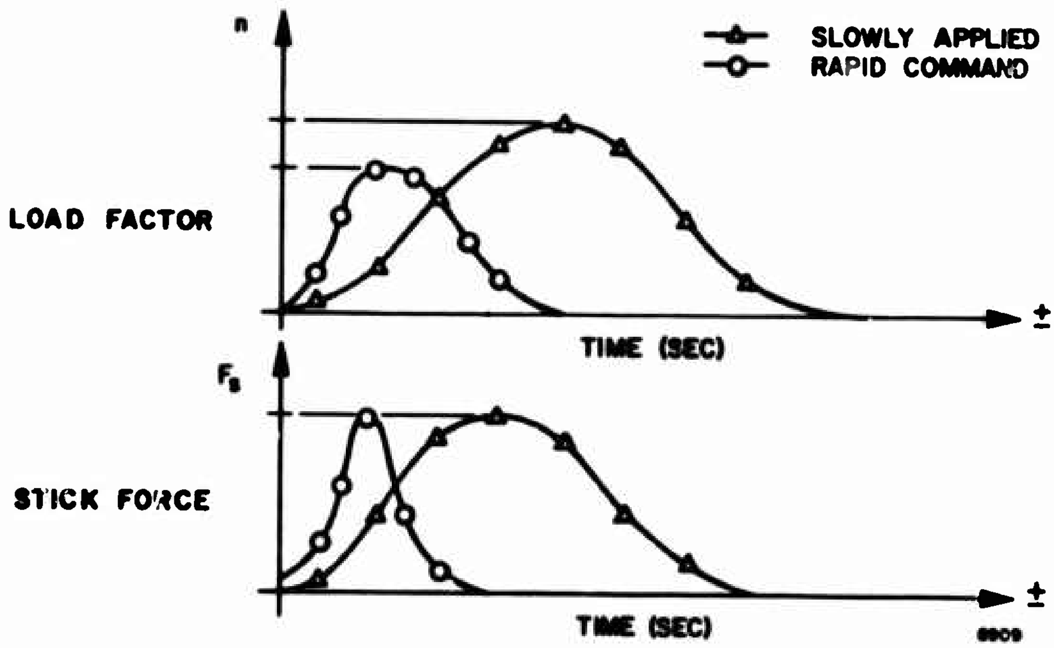


Figure 12
Aircraft Response to Slow and Fast Inputs

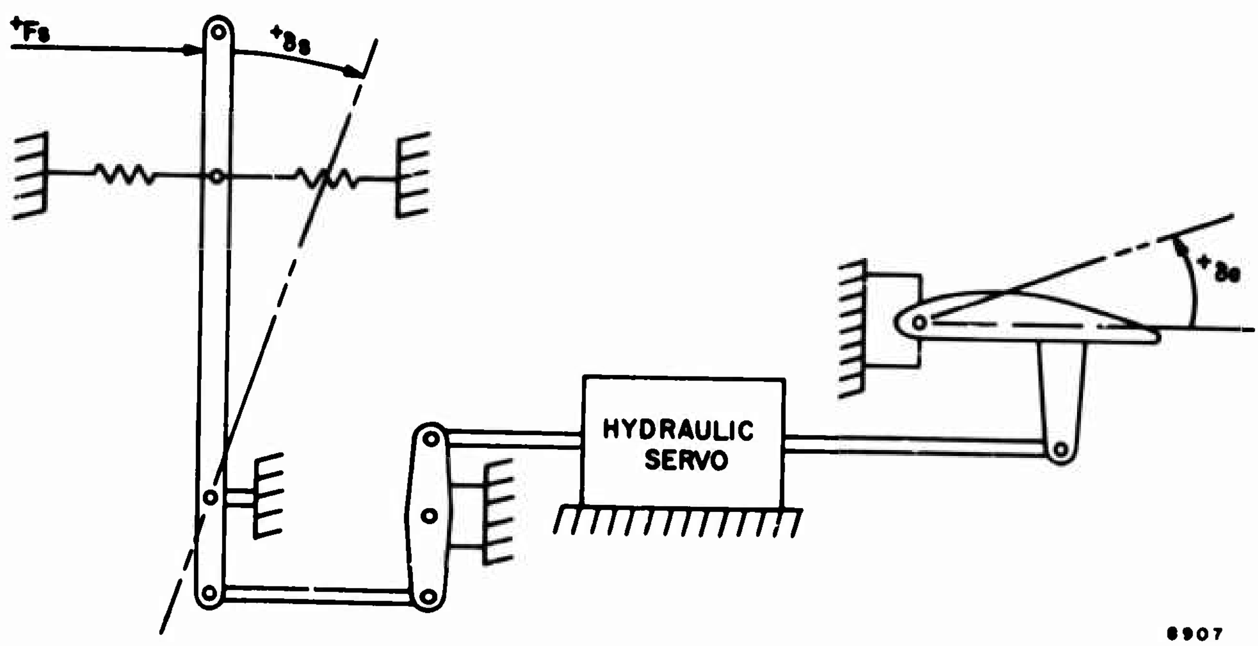


Figure 13
Irreversible Control System

aerodynamic characteristics and making the assumption that $M_q Z_w << u_o M_w$, this expression takes the form:

$$\frac{n_z}{\delta_e} \cong \frac{qS}{m}\left[\left(\frac{C_{M_{\delta_e}}}{C_{M_\alpha}}\right) C_{z_\alpha} - C_{Z_{\delta_e}}\right] \tag{4}$$

The stability derivative C_{M_α} may be further expanded.

$$C_{M_\alpha} = (\bar{X}_{CG} - \bar{N}_o)\, C_{z_\alpha} \tag{5}$$

Introducing this relationship back into equation (4) reduces it to

$$\frac{n_z}{\delta_e} = \frac{qS}{m}\left[- \frac{C_{M_{\delta_e}} C_{z_\alpha}}{(X_{CG} - N_o)\, C_{z_\alpha}} - C_{Z_{\delta_e}}\right] \tag{6}$$

Since the data available is generally in the form of the dimensional derivatives, it is necessary to rewrite equation (6) in the following form:

$$\frac{n_z}{\delta_e} = -Z_{\delta_e} - \frac{I_{yy} M_{\delta_e}}{m(X_{CG} - N_o)\bar{C}} \tag{7}$$

The expression $(X_{CG} - N_o)\bar{C}$ defines, in feet, the difference in the location of the vehicle center of gravity and stick fixed neutral point and I_{yy}/m is required for dimensional correctness. Equations (1) and (7) may now be used as a basis for discussing the handling characteristics of an irreversible control system having a simple spring for artificial feel.

As established in the previous section, the feel requirement is for a constant stick force per increment in normal acceleration. Introducing equation (1) into equation (7) defines an expression for our control system.

$$\frac{F}{n_z} = - \frac{K_o}{Z_{\delta_e} + \dfrac{I_{yy} M_{\delta_e}}{m\bar{C}\,(X_{CG} - N_o)}} \tag{8}$$

Examination of the above equation discloses that at a specific altitude, speed, and center of gravity position, the artificial feel spring constant may be selected to provide the required stick force/g characteristic F/n where $n = (n_z/g)$. However, for a fixed spring constant F_s/n will be a variable: a function of vehicle aerodynamics and the static margin. As an illustration of this point, data were gathered on the Lockheed SST using the above equations. For a given flight condition and static margin, a spring constant K was selected which provided 60 lb/g response characteristic. This

reference system was then perturbed by holding the static margin constant and changing the airspeed by ±15 percent. The results are illustrated by the dashed lines in figure 14. Next, the airspeed was held constant and the center of gravity allowed to deviate from the reference position by ±10 percent. These results are defined by the phantom lines. Reviewing figure 14 and its characteristic equation (8), it is evident that a simple spring artificial feel system is inadequate. The deficiency, as illustrated, is due to the changing aerodynamics and center of gravity movement. Therefore, a simple linear spring will not meet the feel requirements. However, if the spring were augmented by parameters which are related to the change in the aerodynamics and center of gravity location, the resulting artificial feel system would be adequate.

The approach shown in figure 15 utilizes a q-spring and stabilizer trim position to control the feel characteristics. We would like the stick forces to be proportional to the dynamic pressure q and surface deflection; that is,

$$F_s \approx q_{\delta_e} \tag{9}$$

which together with the control surface to stick gearing from equation (1) becomes

$$F_s = K_1 q \delta_s \tag{10}$$

where the original K in equation (1) becomes now $K_1 q$. This method of artificial feel is known in the trade as q feel or $K_q \delta$ feel. The variation of q with velocity, shown in figure 16 is nearly linear up to about Mach 0.7.

A typical q-spring produces a force gradient proportional to the pressure differential across the diaphragm of a bellows. This assumes that the bellows acts as a zero-rate spring or a perfectly extensible membrane. The pressure differential $p_t - p_s$, where p_t is the total pressure and p_s is the static pressure as measured by a pitot tube, can be expressed in terms of the dynamic pressure.

$$q = p_t - p_s = 1/2 \rho U^2 = 0.7 p_s M^2$$

$$q = \text{dynamic pressure lb/ft}^2$$

$$\rho = \text{ambient air density slugs/ft}^3 \tag{11}$$

$$U = \text{true airspeed ft/sec}$$

$$M = \text{Mach number}$$

Then $F = K(p_t - p_s)\,\delta_s = Kq\delta_s = 0.7 p_s\, KM^2\,\delta_s$.

Figure 17 shows the typical control feel responses that can be expected versus Mach number. The q-spring improves the F_s/n characteristics for subsonic Mach numbers. However, for transonic and supersonic Mach

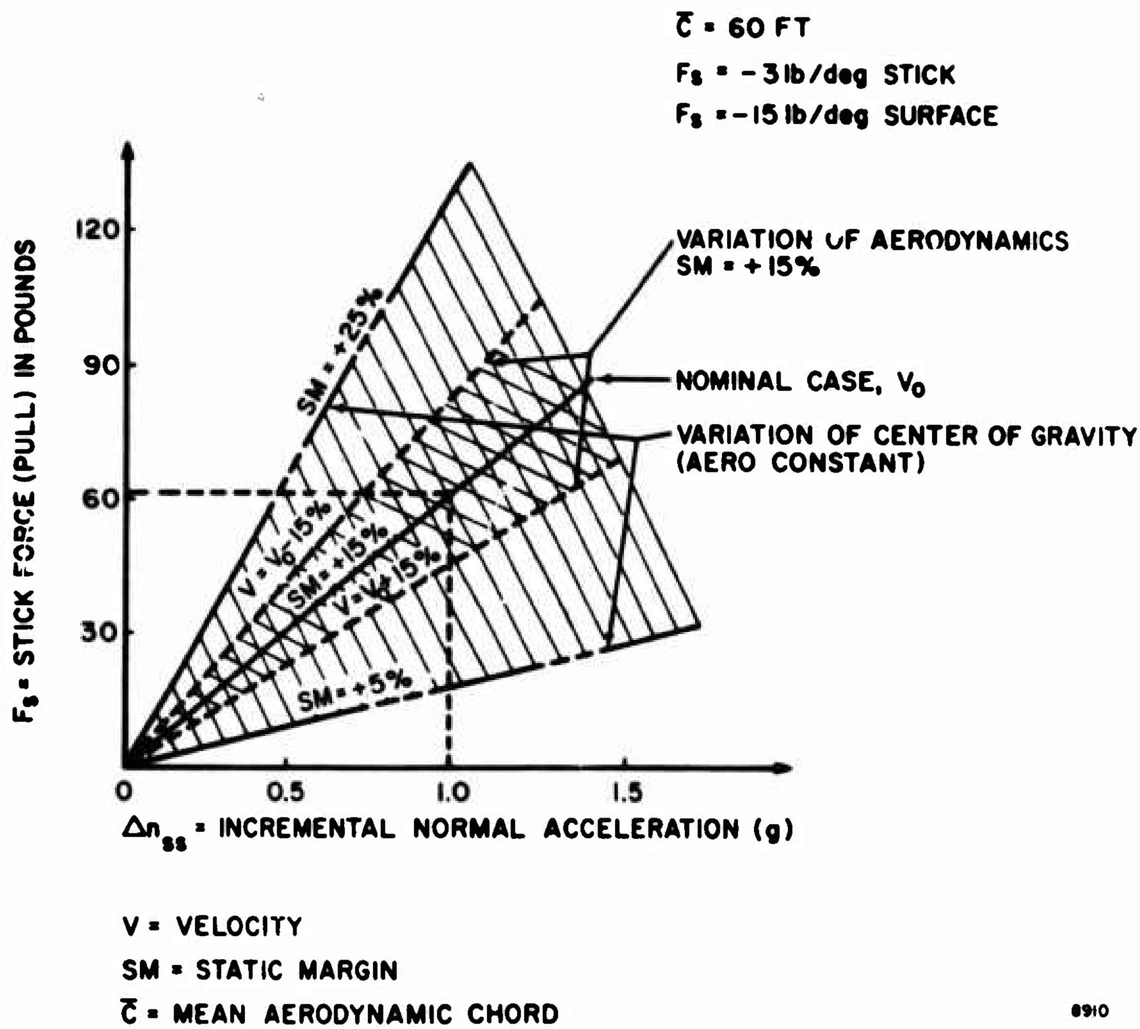


Figure 14
Stick Force Variations

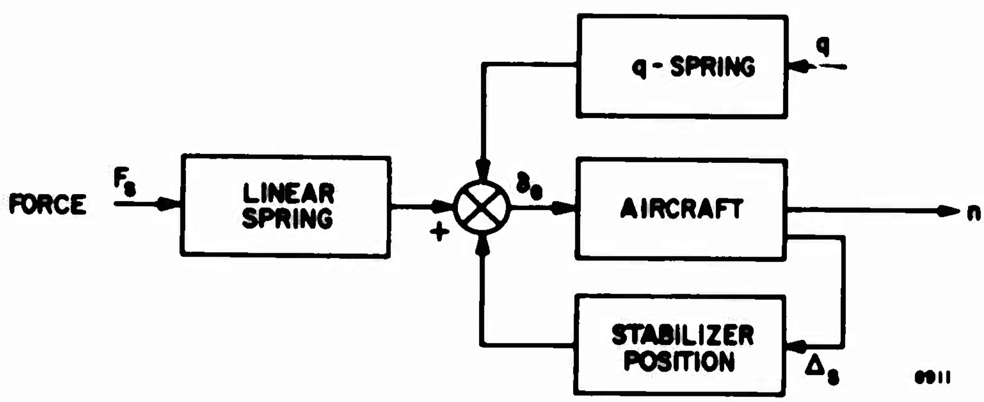


Figure 15
q-Spring Artificial Feel

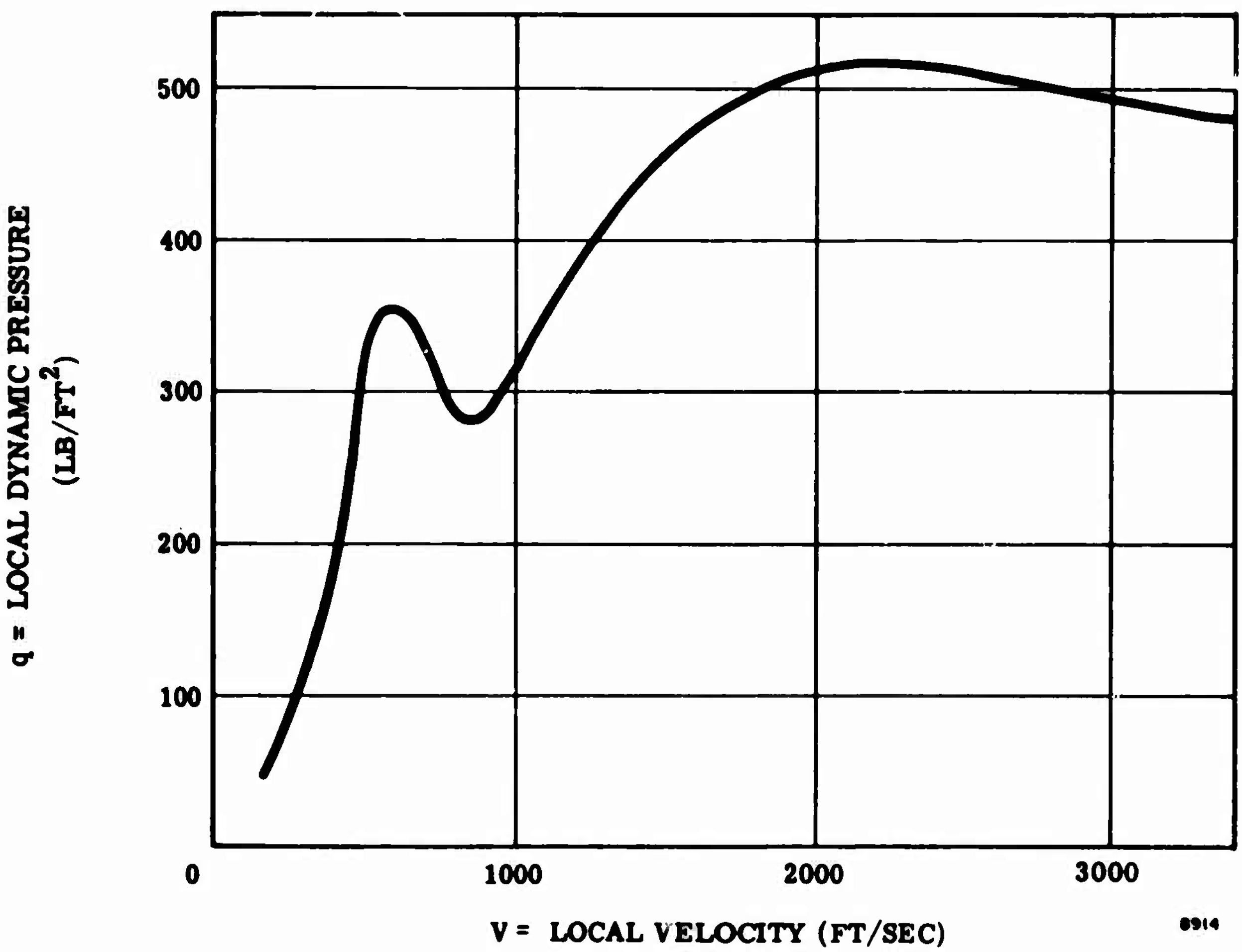


Figure 16
Dynamic Pressure Versus Airspeed

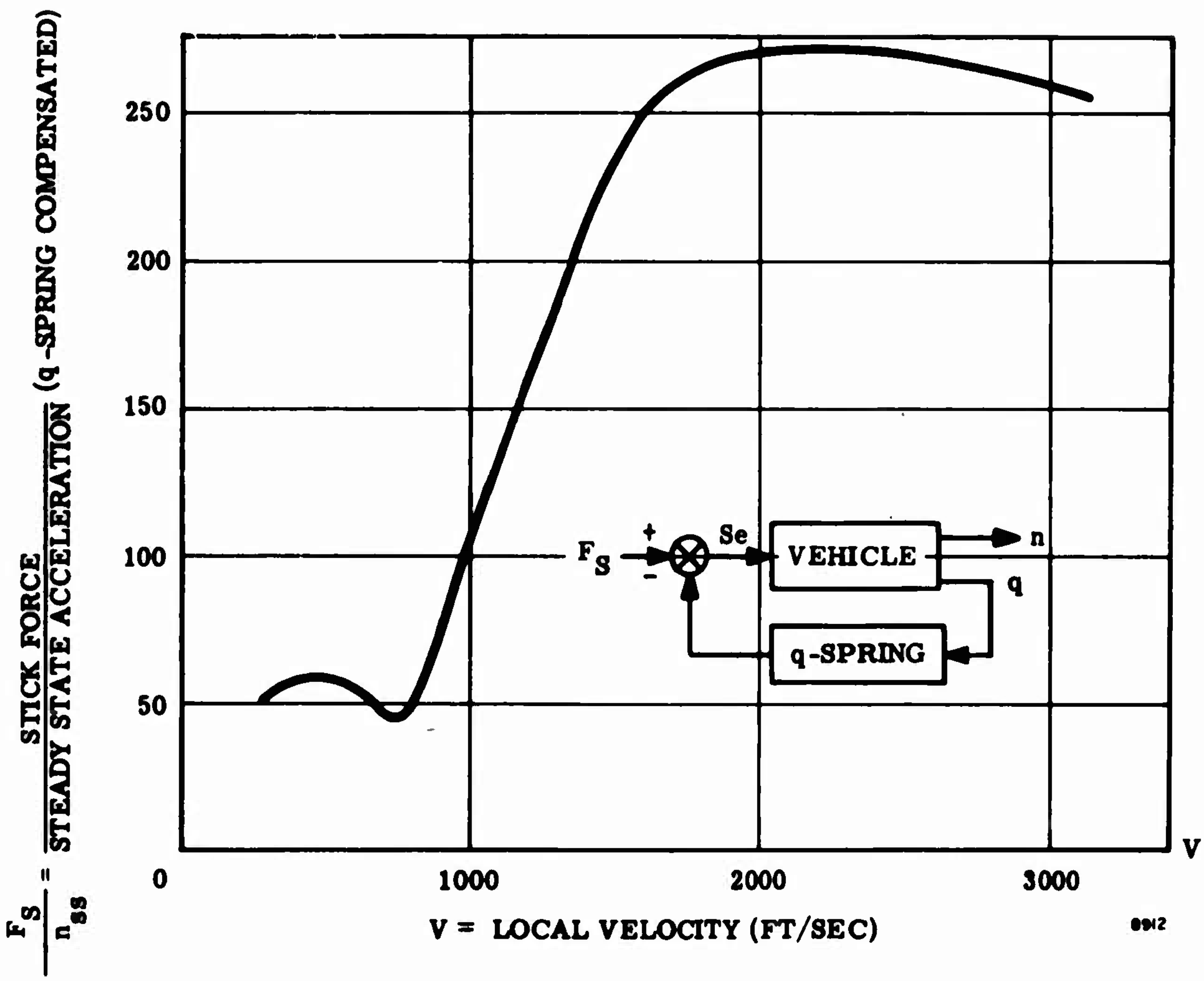


Figure 17
Stick Force For Each Steady-State Acceleration, Lockheed SST

numbers, F_s is too high not only because of the higher q but because of the lower normal acceleration response of the aircraft with increasing Mach number as shown in figure 18. The F_s/n for supersonic Mach numbers can be reduced to a tolerable level by compensating the bellows with springs or air bleed devices, but this adds undesirable complexity. A small electromechanical actuator varies the neutral position of the q-spring according to the trim condition. This accounts for center of gravity and static margin variations and relieves the steady-state stick forces.

Other popular artificial feel producers include single and double bobweights which produce forces in response to aircraft motion. The bobweight amounts to a lead weight cantilevered off the stick or its associated linkage. The spring plus bobweights system produces forces proportional to stick deflection and normal and angular acceleration. The mechanization is simple and reliable, but it is heavy and adds control inertia.

No one has ever designed an optimum artificial feel mechanization, and no one ever will until a standard handling qualities criterion has been established. The feel mechanizations employed thus far in history have been the result of series of compromises of acceptable handling qualities against the various mechanization designs and problems. The situation will be no different for fly-by-wire systems except that the mechanization problems will be much smaller due to the increased design flexibility. One of the goals of fly-by-wire design is to reduce system weight and volume. Therefore, mechanizations using q-springs and bobweights are to be avoided.

The closed-loop method of implementing feel has considerable merit from many aspects including weight, space, performance, and its ready integration with the SAS, CAS, or AFCS for economy of utilization. This method utilizes a feedback blend of normal acceleration, pitch rate, and pitch acceleration because these are the dynamic response cues that the pilot senses. Experience has shown that a pilot attaches less importance to $\dot{\theta}$ as velocity increases since, for a given n_z transient, both the peak and steady-state value of $\dot{\theta}$ are reduced. Conversely, at low velocities as in the approach condition, the $\dot{\theta}$ cue is more important than n_z. Although the blend is not entirely new, it has been defined by Boeing personnel as C* (pronounced "C star") = $K_1 n_z + K_2 \dot{\theta} + K_3 \ddot{\theta}$. C* can be represented by a signal consisting of a blend of the outputs of a pitch rate gyro and a normal accelerometer mounted at the pilot's station. The outputs of the two sensors are combined in a fixed ratio. The relative contributions of each term automatically vary with velocity due to the inherent characteristics of the n_z/δ_e and $\dot{\theta}/\delta_e$ transfer functions. The steady-state relationship between n_z and $\dot{\theta}$ in any aircraft is

$$\dot{\theta}_{ss} \left(\frac{\text{radians}}{\text{sec}}\right) = \frac{n_z \ (\text{ft/sec}^2)}{U \ (\text{ft/sec})}$$

The crossover velocity U_{co}, where the relative contributions are equal, defines the gain of each parameter. U_{co} commonly occurs around 400 fps. The $\ddot{\theta}$ term accounts for the pilot's position relative to the center of gravity.

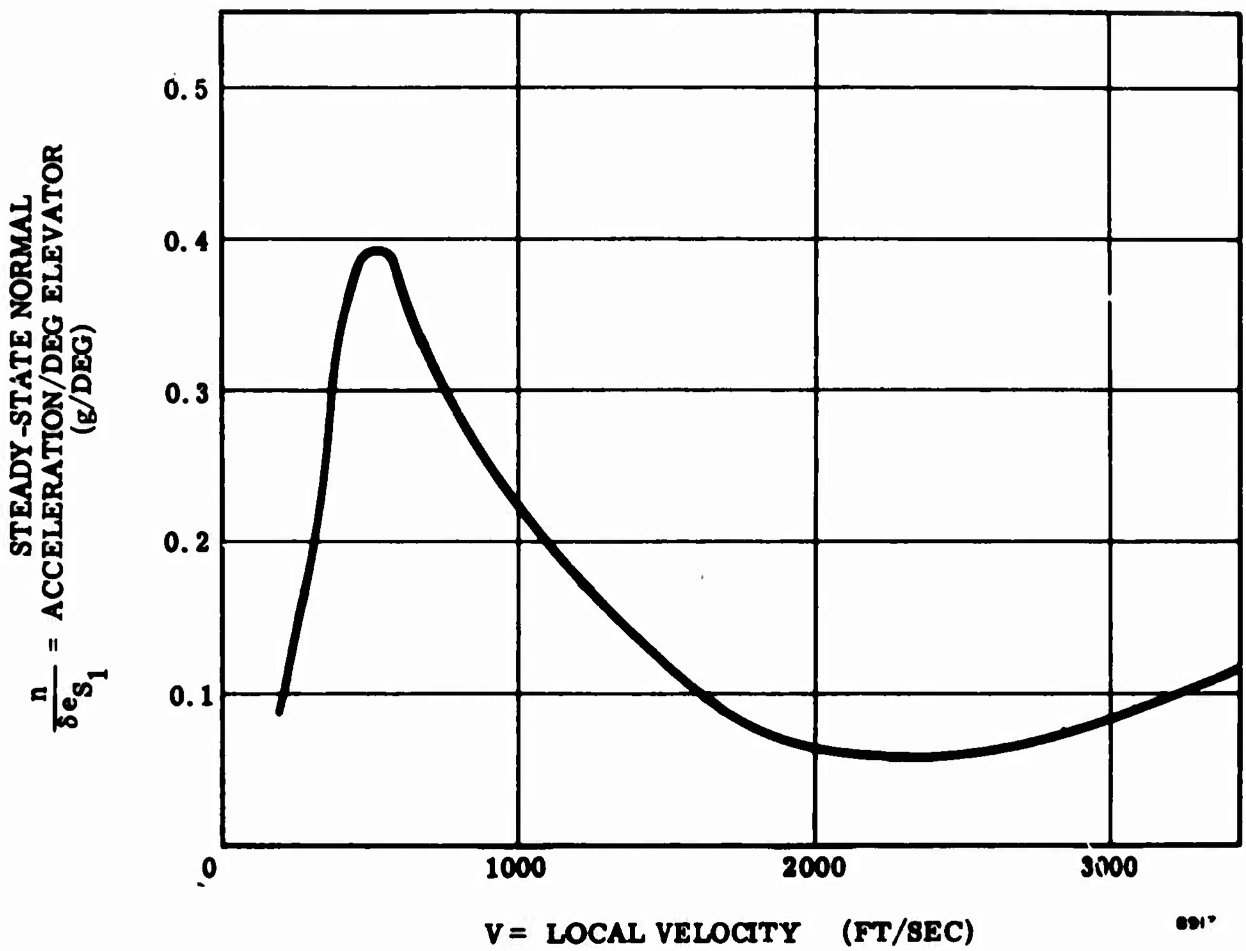


Figure 18
Normal Acceleration Characteristics, Lockheed SST

The F-111 and A-7A use essentially this approach. In the F-111, $C^* = 4n_z + \dot{\theta}$ with the $\ddot{\theta}$ term being incorporated into the n_z term. Pilot acceptance of this scheme is excellent which speaks highly of its potential for fly-by-wire application.

Boeing has proposed the use of the time history of C* for a step command as the new handling qualities criterion to replace the Cornell "thumbprint" currently being used. The thumbprint defines an acceptable area on a graph of short period frequency ω versus short period damping ζ. The argument is that because the pilot senses positions, velocities, and accelerations, and a time history envelope conveys information relating to all of these, a time history envelope is more likely to provide correlation with pilot opinion than the thumbprint. The thumbprint is deemed inadequate because the pilot does not think in terms of ω-ζ, and nonlinear response cannot be properly described.

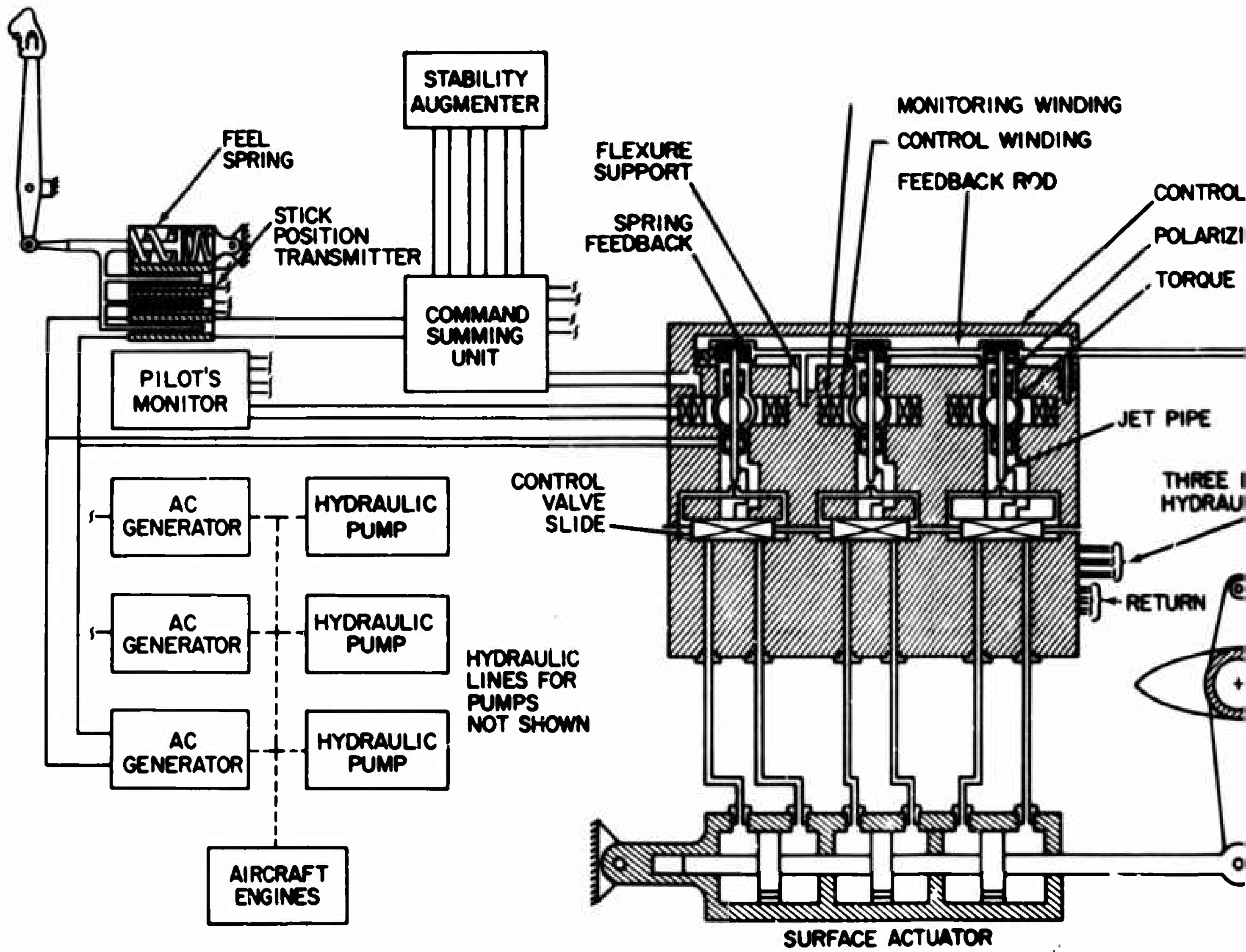
2. PREVIOUS FLY-BY-WIRE WORK

Investigations into fly-by-wire control techniques both in this country and in England date back to the mid-1950's. Unfortunately a major share of the work was done on in-house or classified projects and never found its way into the literature. Most of the reported work has been done on three military funded programs starting in about 1960. Currently two funded programs and at least two in-house programs are known to exist in the United States. A number of fly-by-wire systems have been proposed in the past including the B-70, Concorde SST (France), Avco Vulcan bomber (England), and the Gloster GA-6 fighter (England), but the only system ever constructed was for the X-20 Dynasoar which never flew.

a. Funded Studies

The early funded work started in 1960 when Douglas Aircraft Company, Long Beach, California, was awarded an Air Force contract for study of an Electrical Primary Flight Control System (Ref 9). In 1962 the Army awarded Kaman Aircraft Corporation, Bloomfield, Connecticut, a contract to study Self-Contained Electronic Flight Control Systems (Ref 10) particularly aimed at VTOL aircraft. In 1963 the Army also awarded Massachusetts Institute of Technology, Instrumentation Laboratory, a contract to study "Advanced Flight Control Systems Concepts for VTOL Aircraft" (Ref 11). This last program was more concerned with optimizing flight control than with fly-by-wire.

The Douglas study began in 1960 with the goal of replacing the mechanical flight control linkage between the control stick and the surface actuators with an electrical link in which no electronics or switching is used. The spectre of unreliable vacuum tubes and early transistors very likely spawned the idea of eliminating electronics. Switching was eliminated also for reliability reasons just as it is minimized today. The system operates directly from ship's ac power to eliminate any dc conversion equipment. Therefore, the control stick position transducers are LVDT's (linear variable differential transformer), signal summation uses transformers, and the hydraulic servovalves use ac torquers. Figure 19 shows a diagram of the system for the pitch axis. The system employs triple redundancy to obtain the desired reliability which is equated to the Douglas AD Skyraider pitch



Schematic
Electrical

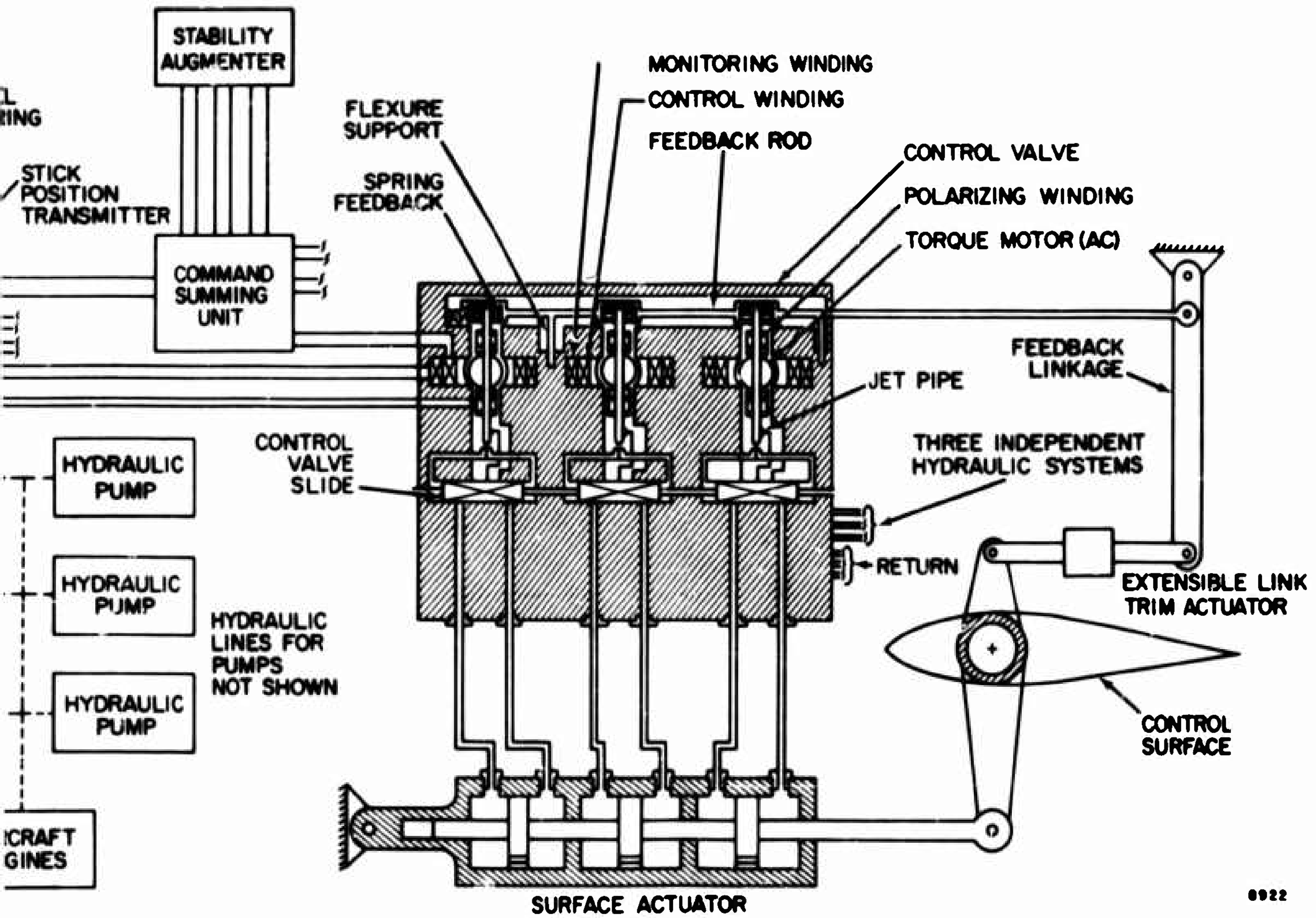


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Figure 19
Schematic Diagram - Douglas
Electrical Flight Control System

control system reliability. Monitoring is performed at the servovalve torquer which also serves as the summing junction for the servo input and mechanical feedback. These are shown schematically in figures 20 and 21. An electromechanical actuator in the actuator's feedback linkage supplies trim. A cockpit display presents the signals from the three servovalve torquers so that the pilot can visually monitor operation of each axis. The signal from each torquer drives one of three small bars on the display. Under normal conditions the three bars move together to form a line that moves up and down. When a channel fails, its bar moves away from the other two. The pilot then notes the difference and disables the failed channel by manually operating a switch that places a choke in series with the electrical signal to reduce the signal to a very low value. This technique is inadequate because the monitor distracts the pilot's attention from his more important flying duties. An automatic failure detection scheme was thereafter devised to eliminate this problem. The scheme compares the torque generated by the servovalve torquer flux against a fixed spring torque. When a failure causes the flux to exceed 105 percent of normal maximum, the spring torque is overcome to operate a hydraulic shutoff valve. This scheme was not implemented in the laboratory model so that neither its effectiveness nor switching time was determined. However, failures in the servovalve second stages would escape detection. The actuator employs three tandem rams and three servovalves having coupled second-stage spools. Active redundancy is employed.

The probability of a failure of the fly-by-wire system was estimated at 3.15×10^{-4} for a 1.5 hour mission compared to 6.15×10^{-4} for the original mechanical system. However, the probability of one failure occurring was 101.7×10^{-4} and 20.1×10^{-4} respectively. In other words, the fly-by-wire system would incur a system failure only half as often, but it would require maintenance actions five times as often as the mechanical system.

While the Douglas study showed that a fly-by-wire system could be designed without electronics or switching to match the reliability of a mechanical system, the study and the design had a number of failings. First, the study failed to include any discussion of artificial feel implementation which is vitally important to a practical fly-by-wire system.

Second, the ac servovalve torquers are very inefficient devices which require a great deal of electrical power from the stick position LVDT for operation, particularly since additional torque is required to operate with the mechanical feedback. The three valves require a total power of 50 watts. The triplex LVDT absorbs another 60 watts at its maximum displacement.

Third, the size and weight of the components are extremely high thus partially negating one of the basic advantages of fly-by-wire of size and weight reduction. The breadboard models of LVDT and servovalve (excluding the actuator) weigh 30 and 55 pounds respectively. Although flightworthy components would certainly weigh much less than this, the trend is obvious. For comparison, a triplex signal LVDT would weigh about 5 ounces.

Fourth, the magnetic summing and monitoring techniques are not practical for two reasons: (1) signals from different power supplies cannot be summed inductively unless they are exactly synchronized; otherwise the

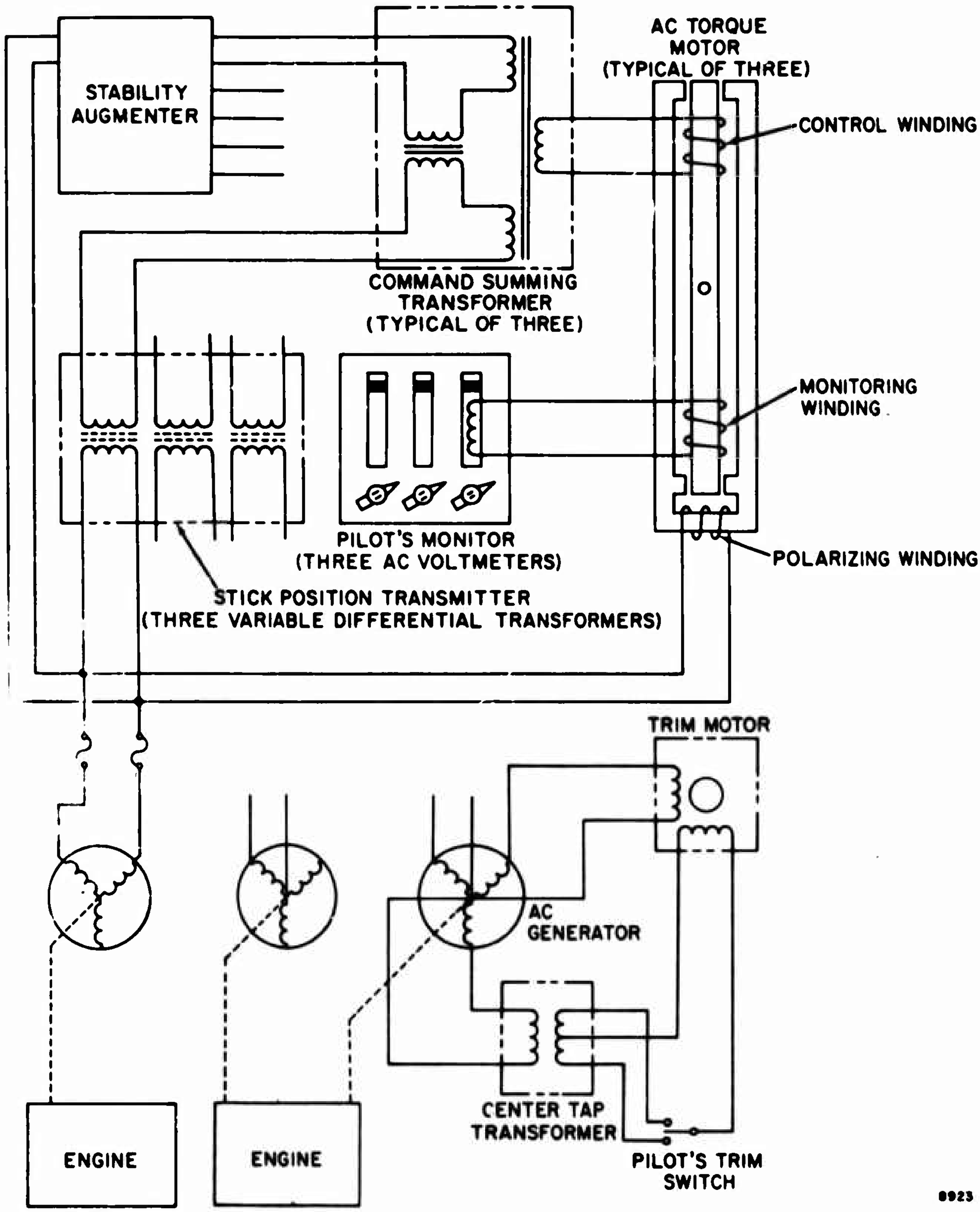


Figure 20
Electrical Schematic - Douglas Electrical Flight Control System

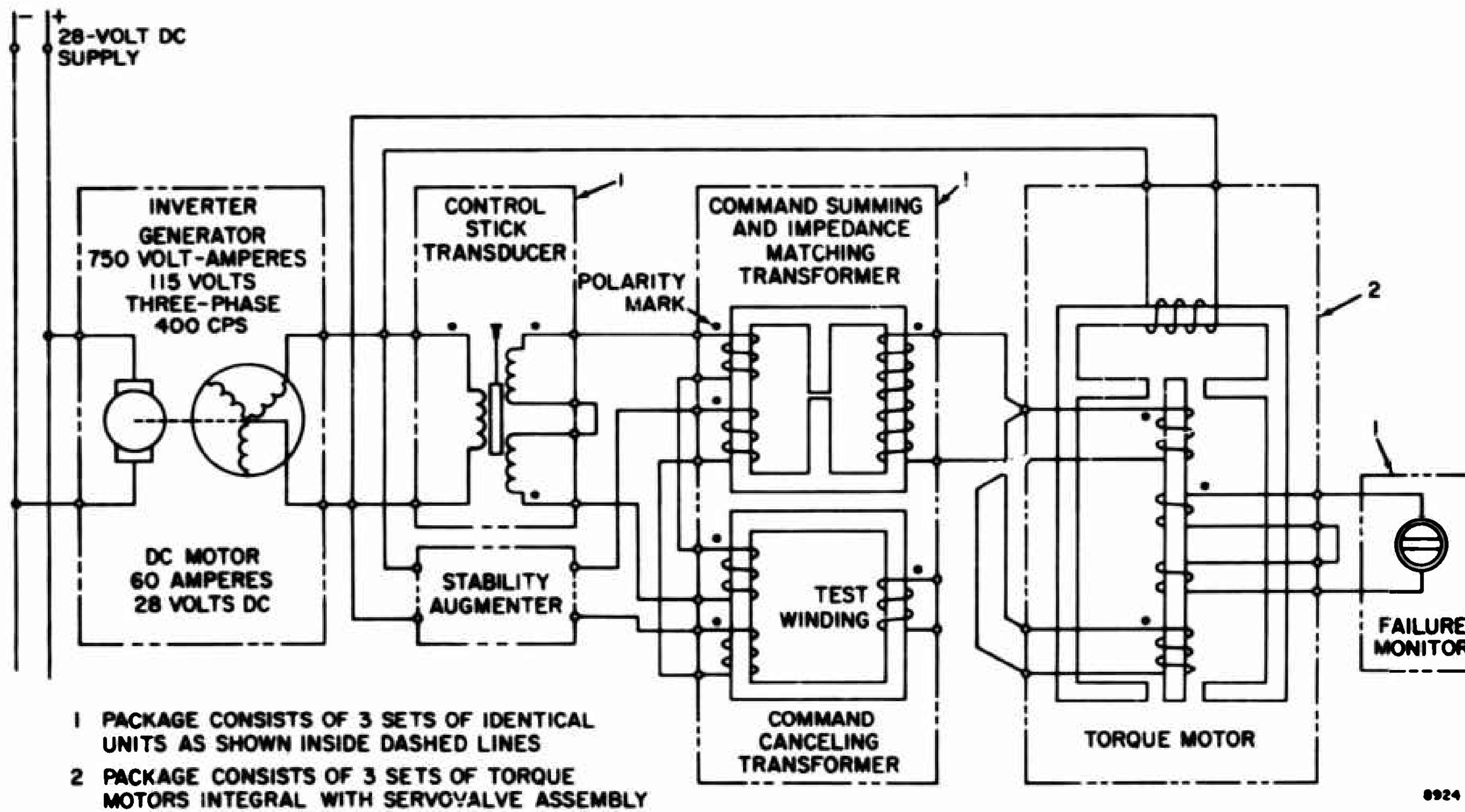


Figure 21
Douglas Electrical Primary Flight Control System - Electrical Circuit Diagram of One Channel (Typical of Three)

output signal will bear no significant relationship to the desired signal; and (2) because the transfer impedance of a transformer depends on the flux level in the core, the output level for one input signal depends on the presence and level of a second input. This nonlinear effect causes a varying forward path gain in the control system.

Fifth, the gradient of surface deflection per control stick displacement is reduced by one-third for each electrical channel failure. One channel failure reduces the command torque at the servovalve input to two-thirds normal which is balanced by the feedback torque produced by two-thirds normal surface deflection. The change in control authority would reduce system performance significantly even for the first failure.

Finally, the use of mechanical feedback and coupled servovalves presents very difficult design and synchronization problems. At least 2 years were spent in developing a prototype model with only limited success. We conclude from the above evaluation that the Douglas approach is not suitable for use in fly-by-wire systems. Although the results were negative, the program has provided a beneficial contribution to fly-by-wire development because it will prevent others from attempting the same approach. Work for the Air Force by Douglas is still continuing but with redirection to include electronics and a different actuator approach.

The Kaman study, which began in 1962, had the purpose of determining whether the intrinsic advantages of self-contained electronic flight control systems (i.e., fly-by-wire systems) could be realized at that time or in the near future while maintaining adequate safety and reliability. A reliability goal was established from failure rate data of the flight control systems of aircraft used by commercial airlines. The goal equals the running average of commercial flight control system failures from 1952 to 1959, which is 0.23 failure for each million flight hours. Equivalent values for military operations were not available. Kaman employed the H-34B twin-rotor helicopter for comparison of fly-by-wire and mechanical systems. Figure 22 shows the flight control installation in that aircraft. Figures 23 and 24 show functional schematics of the derived electronic flight control system (EFCS) for the lateral cyclic and collective pitch axes. The lateral axis is independent of the other three so that the diagram shows the basic technique derived. The collective axis combines the thrust (or lift) and directional axes, and the diagram shows the required interconnections. Figure 23 shows that Kaman has used standby redundancy to achieve a fail-operational system. Triplex induction potentiometers serve as control stick transducers. One transducer provides a reference for comparison with the active transducer. When a failure occurs in either one, the monitor switches out the active unit and switches in the standby one. The system employs dual hydraulic actuators in standby redundancy. The fault detector (monitor) compares the rms values of the command and servo position transducer outputs to detect failures. Upon detecting a failure, the monitor switches out the active actuator and switches in the standby one. The probability of a system failure was calculated as being 10^{-8} for a 10-hour mission. However, this number is not valid for several reasons. First, the generic failure rates were used which assumes that the application K_a is one. However, for the sources quoted, $K_a = 50$ for airborne applications. Hence, the channel failure rate is not 10.66×10^{-6} but rather 533×10^{-6}. This alone brings the probability

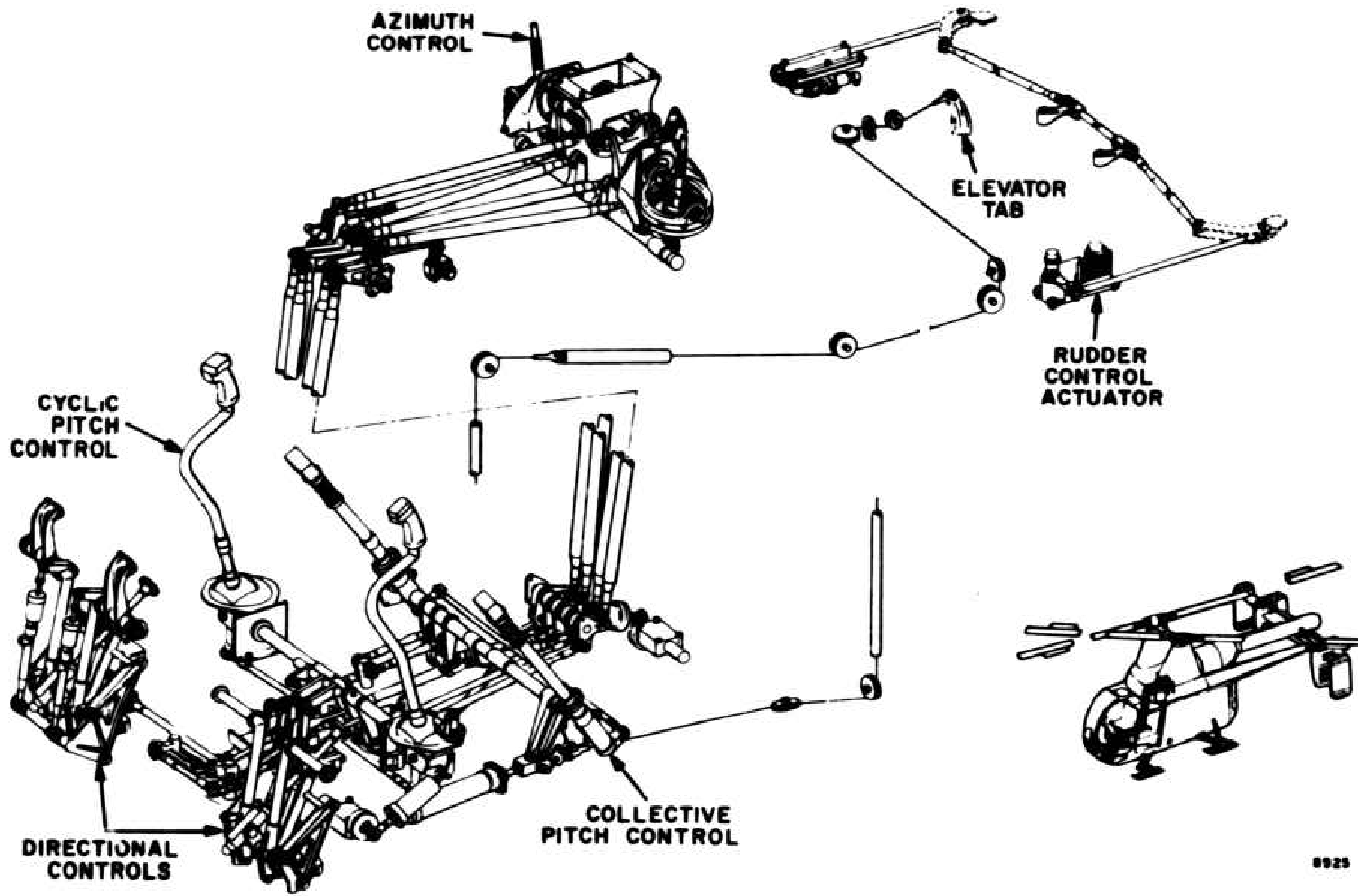


Figure 22
Kaman HH-43B Flight Control System

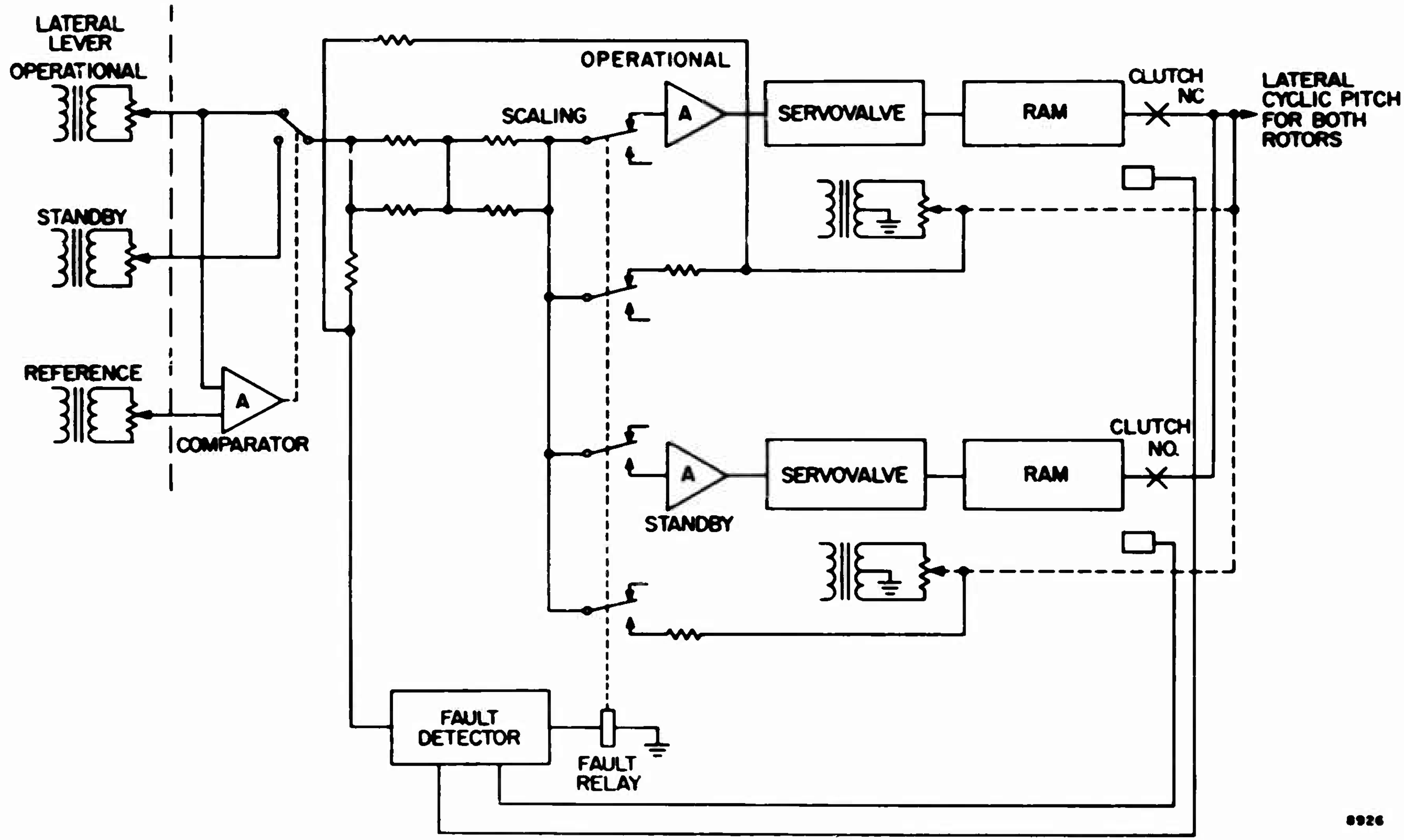
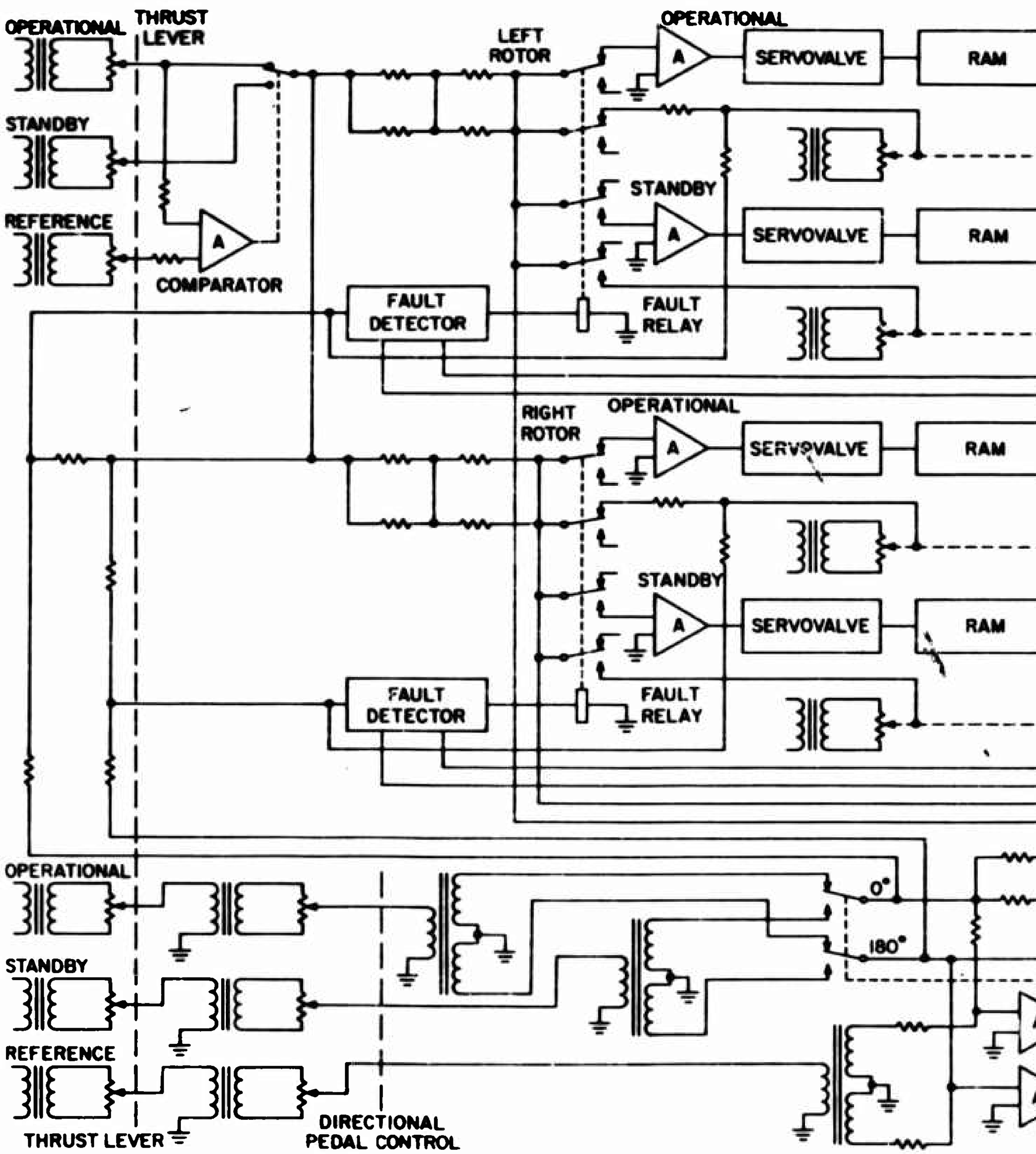


Figure 23
Kaman EFCS Functional Schematic, Lateral Cyclic Axis


OPERATIONAL
THRUST LEVER
STANDBY
REFERENCE
A
COMPARATOR
LEFT ROTOR
OPERATIONAL
A
SERVOVALVE
RAM
STANDBY
A
SERVOVALVE
RAM
FAULT DETECTOR
FAULT RELAY
RIGHT ROTOR
OPERATIONAL
A
SERVOVALVE
RAM
STANDBY
A
SERVOVALVE
RAM
FAULT DETECTOR
FAULT RELAY
OPERATIONAL
STANDBY
REFERENCE
THRUST LEVER
DIRECTIONAL PEDAL CONTROL
0°
180°
A
A

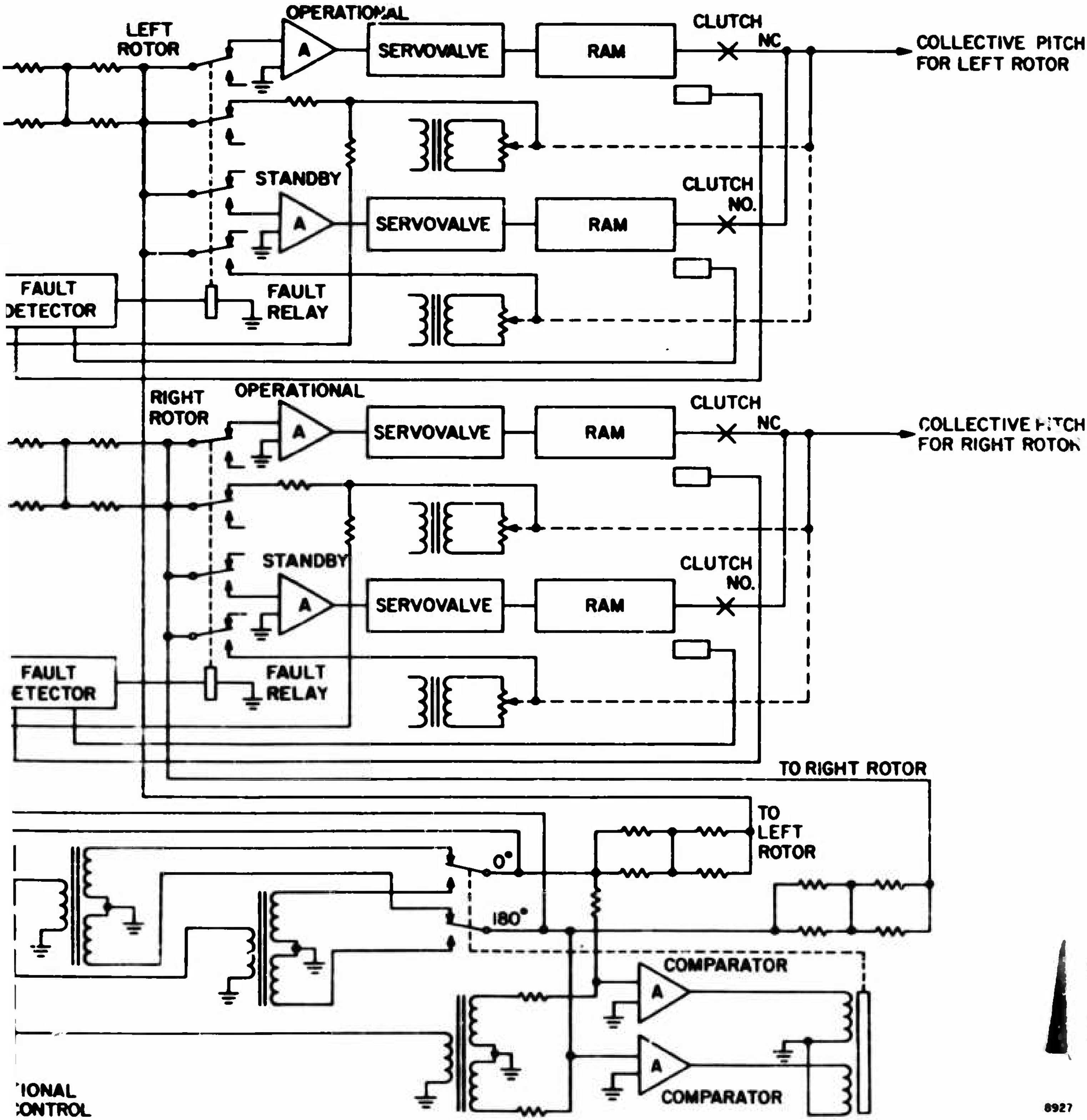


Figure 24
Kaman EFCS Functional Schematic,
Collective Pitch Axis

of system failure up to $(10 \times 533 \times 10^{-6})^2 = 28.4 \times 10^{-6}$. Further, a number of single elements can fail that will cause system failure (i.e., in series with respect to reliability). These include the monitors, clutches, and relays. Also, since the standby channels are not monitored, the monitor could switch in a failed channel. Assuming an optimistic failure rate of 10^{-5} for each of the four series elements (two clutches and two monitors and relays), the probability of system failure becomes $4 \times 10 \times 10^{-5} + 28.4 \times 10^{-6} = 4.28 \times 10^{-4}$ for a 10-hour mission for each of the four axes for a total system rate of 1.8×10^{-3}. This more realistic value is a long way from the goal of 0.23×10^{-6}. The results of this system design study point up the need for eliminating the series reliability elements, monitoring all of the system elements, and employing enough channels to maintain operation after two failures.

The MIT study, which began in 1963 and is still in progress, is to develop advanced flight control systems for VTOL aircraft. The objective is to develop practical control systems which provide optimum control characteristics for VTOL aircraft throughout their flight regime under all weather and combat conditions. This program considers the manual and automatic flight control systems as an integrated system to provide the optimum system. The study concerns only the functional aspects of the control system; it does not consider such factors as reliability, maintainability, cost, and weight.

The MIT and Sperry Phoenix programs are approaching the problem of optimizing the design of aircraft control systems from opposite directions. Yet they are arriving at very similar conclusions. The MIT approach, in the process of determining the optimum controller configuration, has determined that incorporating the fly-by-wire approach is desirable. The Sperry Phoenix approach, in the process of determining the optimum fly-by-wire system configuration, has determined that incorporating artificial feel (i.e., the controller) is desirable. The fact that the MIT study is limited to VTOL aircraft does not alter the conclusions.

MIT is currently flight testing their concepts in a Vertol CH-46C in which the copilot's mechanical system has been replaced by an electrical link and the advanced flight control system. The pilot's mechanical controls remain in the airplane for backup since the advanced system is nonredundant. Therefore, in the strict sense, the system is not fly-by-wire at this time; it would be classified as a pseudo fly-by-wire system because it has mechanical reversion capabilities. The exact system implementation is not known except that the electrical system drives through the mechanical system so that the safety pilot's controls move in parallel with the electrical stick inputs. Position transducers are employed on the copilot's control sticks to generate the electrical command signals. The controller, which is comparable to the artificial feel system, consists of an inertial velocity measurement system for flight path control during hover and cruise.

b. In-House Studies

In addition to the funded studies, a number of in-house programs are known to have existed or are presently under way. Very little is known about the results of these works since they seldom find their way into the literature. Discovery of the existence of such programs comes during plant

visits or private conversations. In-house fly-by-wire studies are known to have been done at various times at General Dynamics/Convair (San Diego) and Fort Worth, Minneapolis-Honeywell, North American Aviation (Los Angeles), Boeing (Seattle), Sud Aviation (France), Avco (England) and Elliott Brothers (England). In-house studies of various levels are now under way at the Flight Dynamics Laboratory of the Research and Technology Division (Wright-Patterson Air Force Base) and Vertol (Morton, Pennsylvania).

The Convair study (Ref 12), performed in 1956-7, was one of the earliest reported works in fly-by-wire control. Convair recognized the problems of mechanical control system designs particularly in high-performance interceptor aircraft. They also recognized the benefits of a closed-loop control system using aircraft rate feedback to obtain the desirable handling characteristics (constant stick force per g and positive trim stability). A moving cockpit simulator having a single degree of freedom (pitch axis) allowed evaluation of their concepts. Both center and side sticks were available for evaluation. System evaluations were performed by giving the pilot a task of tracking a target or holding a specific attitude while simulated gust disturbances were being introduced. The conclusions of the study were that a fly-by-wire could be designed having as good a performance as a mechanical system, and it would also be lighter and more flexible. Further, the system should use ac transducers and active triple redundancy for improved reliability. Although the confidence level of achieving adequate system reliability at that time was very low, the confidence in the future application of fly-by-wire was very high.

The more recent work at General Dynamics/Fort Worth investigated the application of fly-by-wire to the F-111 to reduce weight and save space. While the weight and volume were reduced approximately in half, the question of proven reliability prevented its application except for the spoilers. These are discussed later under Applications and described in Section VIII.

In 1958-9 Minneapolis-Honeywell studied the application of fly-by-wire concepts to future supersonic aircraft (Ref 13). This theoretical study also concluded that fly-by-wire systems held considerable promise in solving the growing problems of mechanical systems -- if only the reliability could be improved to match that of the mechanical system. The proposed solution was to use a fail-operational primary system with a simple standby channel for emergency backup. Again, a closed-loop control system was used by employing rate and/or acceleration feedback, but surface rate was also added as a feedback parameter for integral control. The liquid metal servovalve[1] being developed by General Electric (Ref 14) for the Air Force was proposed because it has no moving parts and should be, therefore, very reliable. This method uses the eutectic alloy of sodium-potassium-cesium which remains liquid from -102°F (-74.4°C) to +1332°F (+722°C). Because a conductor carrying a current in a magnetic field develops a force, the liquid can be pumped by an electromagnetic input to form a servovalve. The command inputs, electronics, and feedback sensors are triplex and fail-operational. The actuator is dual but with no monitoring specified. Design recommendations include separately routed cables using wire with mixed steel and copper strands, transformer

[1]Development of NaKCe components is still underway with the first flight test scheduled for 1969.

isolation to eliminate the effects of shorts, the use of inductive transducers, and reducing the number of connections wherever possible.

The B-70 Division of North American Aviation (Los Angeles) investigated the application of fly-by-wire techniques to the XB-70 control system. This airplane has a very difficult design problem because the cable runs are very long and routing is difficult. This was discussed previously. The design study did not proceed very far when reliability uncertainty squelched the project. The throttle system remained electrical, however, because an operational mechanical linkage could not be designed.

Little is known of the other past in-house efforts. Sud Aviation originally designed the Concorde supersonic transport to use fly-by-wire control, but the Federal Aeronautics Agency demanded mechanical reversion capability before it would allow the aircraft to be used by the U.S. carriers. Avco (England) at one time proposed a fly-by-wire conversion of the Vulcan bomber, but it was not accepted. Elliott Brothers fabricated a highly redundant cockpit mockup of a fly-by-wire system for exhibit and demonstration at an Aviation Exposition in England in 1963.

The Flight Dynamics Laboratory (FDL) in-house program, which began in 1966, is aimed at converting a B-47 to fly-by-wire to provide flying proof of its feasibility. The conversion will progress in stages beginning in 1967 with a nonredundant pitch axis system having mechanical reversion. Flight tests will provide checks on the performance characteristics. A C* feedback system will then be installed in conjunction with a side-stick controller. The side-stick controller and C* feedback will then be used with a redundant servo actuator using hydraulic logic. The final phase of the in-house program will be test of a liquid metal actuator package with C* feedback and a side-stick controller. The Sperry Phoenix program is related to and operates on a parallel timetable with the FDL program. Sperry is tentatively scheduled to fly a redundant three-axis system in 1969 in a second B-47.

The Vertol in-house study program is in the preliminary stages of planning. The program is concentrating on applications of fly-by-wire techniques to future VTOL aircraft of all types and sizes including rotary and tilt wing. Vertol refers to the electrical flight control system which replaces the mechanical system as an Advanced Flight Control Linkage (AFCL) rather than a fly-by-wire system.[2] Preliminary planning calls for design, fabrication, laboratory test, and flight test of a prototype system in a CH-46 helicopter. Sperry has recommended a fly-by-wire implementation for the AFCL based on the results of the present study program. The system is described in Section VIII.

c. Applications

While the applications of fly-by-wire technology to aircraft are very limited, other applications range from submarines to space vehicles. Known

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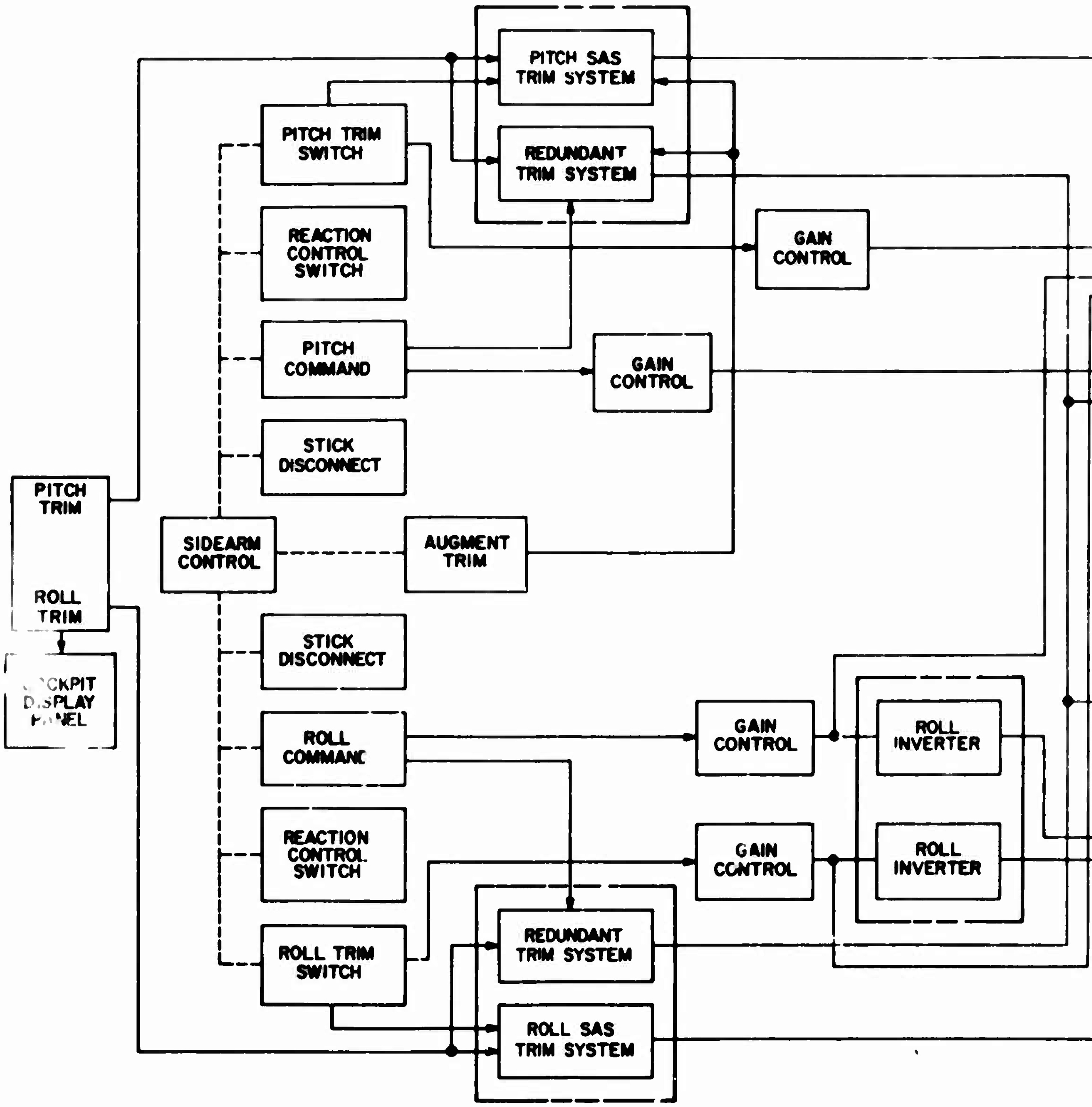
[2]Very likely the study has been strongly influenced by the MIT program which is using a Vertol CH-46 machine for flight test work. MIT refers to their optimum controller as an Advanced Flight Control System.

applications include the X-20 Dynasoar, the Mercury-Gemini capsule series, Apollo and LEM (lunar excursion module), F-111 spoilers, and XB-70 and CL-44 Argus throttles. The throttle system problems were discussed earlier. Several applications have been proposed and denied as we have already mentioned. A proposed English fighter, the Gloster GA-6, was to have had a fly-by-wire system, but the entire program was cancelled at an early stage. No information has been obtained on this aircraft.

A little-known fact in the aerospace industry is that the newer submarines use fly-by-wire control or, in some cases, fly-by-fluid control. This fact is not so surprising when we stop to consider that the control mechanization problems are not much different in submarines than in aircraft. They are also subject to friction, deadzones, compliance, hysteresis, backlash, routing problems, and body bending. Their control frequencies are several orders of magnitude lower, but they are "flown" through the water by an operator who controls pitch, roll, and yaw control surfaces much like an aircraft. At least one hydrofoil craft was known to have used electrical control linkages from the cockpit to the foil actuators.

The X-20 Dynasoar flight control system (Ref 15) is the only known existing fly-by-wire system, yet even this system never flew. The X-20 simulator, which uses most of the prototype hardware, is located at the Flight Dynamics Laboratory of the Research and Technology Division, Wright-Patterson Air Force Base. Figure 25 shows a block diagram of the elevon control system. Most of the details of the aircraft are still classified, but conventional automatic flight control system design techniques were employed. The significant points for the purpose of this study are as follows. The primary control system is fail-operational with an additional direct electrical link available for emergency backup control. The system is functionally very similar to the F-111 flight control system in that it uses essentially C* feedback and an adaptive gain control loop to maintain maximum servo gain and optimum handling characteristics. While the adaptive gain control is triplex in both systems, the remainder of the X-20 system differs in that it is only duplex. In-line monitoring and hardover detectors continuously and independently check each channel to achieve the fail-operational capability. In case both channels fail, a direct link is available to provide a fixed surface deflection per stick deflection gradient. The backup link has no artificial feel, of course, and operators find flying it through the transonic range almost impossible. The probability of a system failure for a 1-hour flight is estimated to be 3×10^{-6} which is still an order of magnitude higher than the Kaman criterion of 2.3×10^{-7} for a 1-hour flight (commercially). Since presumably the 3×10^{-6} figure is acceptable, the ratio of 10:1 could establish the criterion for military flight control systems safety with respect to commercial flight safety.

The vehicles of the various space programs, Mercury, Gemini, Apollo, and LEM, use fly-by-wire techniques to save weight and space. The Mercury system was actually a pseudo fly-by-wire system since it had a mechanical backup system. The surprising fact about these systems is that they tend to use nonredundant channels and stress the alternate modes approach to redundancy rather than replicating channels. For example, the LEM has three alternate modes of control varying degrees of degraded performance from the


PITCH SAS
TRIM SYSTEM
REDUNDANT
TRIM SYSTEM
PITCH TRIM
SWITCH
REACTION
CONTROL
SWITCH
GAIN
CONTROL
PITCH
COMMAND
GAIN
CONTROL
STICK
DISCONNECT
PITCH
TRIM
ROLL
TRIM
SIDEARM
CONTROL
AUGMENT
TRIM
STICK
DISCONNECT
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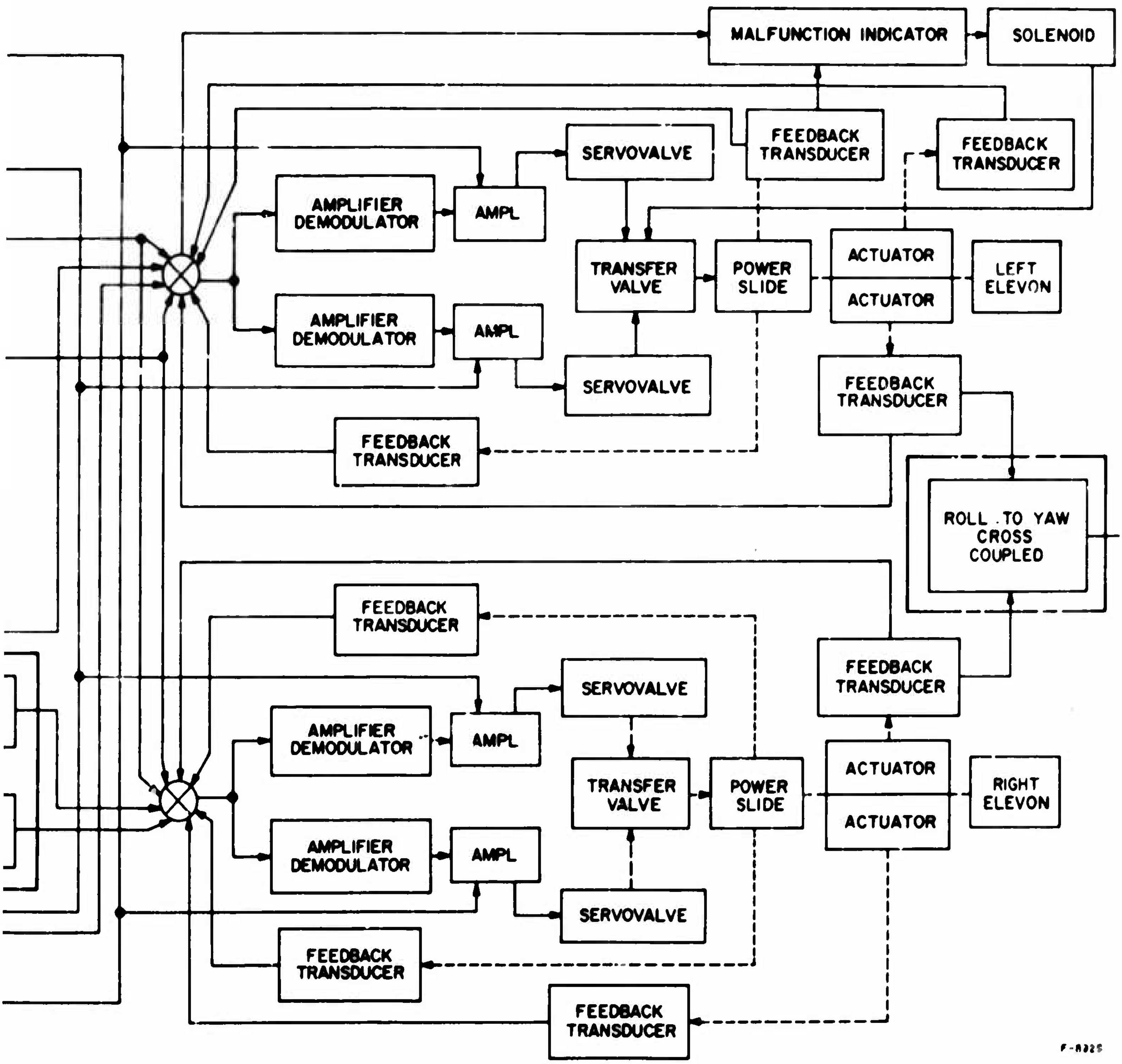

Figure 25
X-20 Elevon Control System
Block Diagram

primary mode. (The X-20 had two alternate modes although replication was used.) By using well-tested, high-reliability components (such as Minuteman quality), single-channel reliability can be made very high for these relatively simple systems. Pilot monitoring further simplifies them. The adequacy of pilot monitoring for space vehicles was borne out when Gemini experienced a primary roll axis hardover failure during the docking maneuver with the Agena target vehicle. The failure was isolated, the primary axis deactivated, and the secondary system activated within a matter of a few seconds. This type of operation is allowed because such limited tumbling motions generally cause no harm as long as the pilots do not become disoriented. Little danger of collision exists except possibly while docking.

The F-111 spoilers are fly-by-wire for two reasons. First, it is required because of the difficult design problem presented by the swing wing; and second, it is allowed because the spoilers are secondary roll control devices which are active when the wing sweep is less than 45 degrees. The rolling tail furnishes the primary roll control. Performance of this system is being watched with interest because it is the first semblance of fly-by-wire in an operational aircraft. Over 2,000 flight hours have accumulated on these spoilers as of this writing without a system failure. Component failures have occurred in the servos. They have been caused either by faulty manufacture or by the incompatibility of some of the exotic metals being utilized, namely titanium bearing on titanium. These problems reportedly have been eliminated.

The problem posed by the swing wing is that the hinge point for the spoiler control linkages varies with wing sweep angle. A very complex mechanical arrangement would be required to accommodate these variations. Therefore, electrical linkages have been implemented to solve the problem. Dual redundancy is employed since two sets of spoilers are used on each wing. A nonredundant actuator and channel operates each spoiler. When one spoiler fails, it is locked down along with its mate on the other wing to maintain symmetry.

One other application of fly-by-wire which may come about soon is on the lift fan VTOL in which the thrust diverter louvers, particularly in the wing, are very difficult to control mechanically. This problem occurs in the VS/FRG Mach 2 VTOL Fighter.

3. RELATED WORK

In addition to the known fly-by-wire studies which have been discussed in the preceding sections, a number of other programs and systems exist that are strongly related to fly-by-wire systems design and technology. Such programs included command or control augmentation systems, fully boosted control systems, model reference flying simulators and redundant flight controls.

Command augmentation is a technique of paralleling the mechanical control command with an electrical stick command signal into the stability augmentation system. The electrical signal bypasses the friction and nonlinearities of the mechanical system for improved control. This technique is also called control stick steering. The stick transducer may be either position or force; both are used. The signal can be shaped to provide the desired rate response from the aircraft since a command or control augmentation system is effectively a limited-authority (because the SAS normally is authority limited) rate autopilot. The resulting control system provides more accurate rate control. With high SAS gains, the system provides a very high static stiffness to the control system producing an effect much like integral control. Normal acceleration feedback is then added to achieve better response at high speeds where the rate response drops off. The rate loop then dominates control at low speeds where the acceleration response drops off. The addition of acceleration feedback also eliminates dependence on air data for gain control. With the addition of acceleration feedback, the CAS becomes more than a rate autopilot. It becomes essentially what has been called a C* command system in parallel with the mechanical control system. Gain control of the inner stability augmentation loops may be either fixed or variable through adaptive controllers. The system provides heavy gust damping while providing fast, well-controlled responses to commands. The resulting CAS is a full-time system which becomes the primary control system with the mechanical system being used as a backup. This is the approach being used on the A-7A (Ref 16) with fixed gains and on the F-111 (Ref 17) with adaptive gains. Pilot comments on the handling qualities of these systems are very favorable. The approach will also be utilized in the newer high-performance aircraft such as the SST, Boeing 747, advanced manned strategic bomber (AMSA), and the advanced fighter/attack aircraft (VFAX/FX).

The importance of CAS to fly-by-wire technology should be obvious from its functional similarity. Removal of the mechanical backup controls from a CAS leaves a system very similar to fly-by-wire.

Fully-boosted control systems are of interest primarily because they are irreversible and require artificial feel as does fly-by-wire. The problem of implementing artificial feel or in optimizing the controller is common ground which we have already discussed. The past and continuing work being done in this area is directly applicable to fly-by-wire technology.

Another facet of boosted systems of interest is their mechanical reversion capability. Mechanical reversion allows the pilot time to recover control while correcting or clearing a fault (if possible). Runaway or hardover controls are the primary reason for mechanical reversion rather than power failure. The lack of mechanical reversion is the biggest deterent to the application of fly-by-wire control today. Yet the Caravelle commercial transport has no mechanical reversion. It relies on triplex hydraulic supplies and actuators to provide necessary reliability. This aircraft has supplied an important lesson in fly-by-wire design: route the separate control and power lines as far apart as possible. A Caravelle was lost because the design violates this rule. All of the aileron hydraulic lines pass

through the wheel wells. A brake fire in one of the wheel wells burned through all of the hydraulic lines causing loss of lateral control and the airplane. Hopefully, this problem has been eliminated.

A flying simulator is an aircraft modified by the addition of an electrical flight control system in which the dynamic characteristics of the aircraft can be modified through an electronic model for study of stability or handling qualities or for study of the flight characteristics of proposed aircraft. The original flight control system is retained for normal flying and for safety since the electrical system is always nonredundant. The two systems are functionally independent up to the point where the simulated system's output sums into the original system which usually occurs at a series actuator. The electrical system is functionally related to fly-by-wire systems in that it employs various electrical sticks and it can vary and control the handling qualities of the airplane. The USAF and NASA have modified a number of fighter and small bomber aircraft (such as the B-25, F-94, F-101, and F-102) for variable stability studies, to evaluate sidestick controllers and adaptive flight control systems, and to simulate such aerospace vehicles as the X-15 and X-20 for pilot training. We have already mentioned the converted CH-46 that MIT is using for advanced flight control system studies. Boeing (Seattle) has converted their 707 prototype, the Model 367-80, for variable stability and handling qualities studies. Grumman is building for the Navy seven copies of the TC-4C, which is a modified Gulfstream I, that will simulate the A-6A for flight training. A complete A-6A cockpit will be constructed in the passenger compartment, and all of the A-6A avionics will be included in the airplane. Cornell is modifying a Convair C-131 (a turboprop version of the C-340) for a total in-flight simulation of such advanced aircraft as the AMSA, SST, and C-5A. An entire cockpit and nose section will be added forward of the existing cockpit which will be retained. The added section will be changeable to allow simulation of the various aircraft. Variable stability flying and simulation will be done by using response-feedback techniques as well as by model-reference techniques using an on-board computer.

The application of redundancy to flight control system design is of particular interest to fly-by-wire technology because through its application, the required system reliability and safety result. Redundancy has been applied primarily to those subsystems of an aircraft affecting flight safety such as the SAS, CAS, or the AFCS in all-weather landing (AWL) modes. The degree of applied redundancy relates to the degree that the system affects flight safety. The SAS or CAS in most high performance aircraft is now or will be triplex. Such aircraft include the B-58, F-111, X-15, X-20, SST, 747, C-5A, AAFSS, and IHAS. When Category III[3] AWL comes to pass, the AFCS will very likely be triplex. These systems are mostly required to be fail operational. Triple redundancy with voting is the brute force technique of gaining that end; it is inefficient and adds undue complexity, cost, and

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3 Category III has several subclasses, but it essentially refers to zero visibility conditions.

weight to the system. Fly-by-wire systems require a greater failure tolerance since they must generally operate after double failures to obtain the desired degree of reliability and safety. This means either employing a higher degree of redundancy, using more finesse in applying redundancy to optimize it, or both. Sperry Phoenix has had a program for several years which uses finesse in optimizing redundancy. The technique, called fail-passive design (Ref 18), designs out the causes of active (i.e., hardover) failures so that the resulting channels or components fail in a passive manner only. A fail-passive component or channel fails in such a way that it has no output and does not interfere with the normal operation of a parallel component or channel. By using fail-passive design, a fail-safe system requires only one channel not two, a fail-operational system requires only two channels not three, and so on. Furthermore, little or no monitoring or switching equipment is necessary.

To demonstrate the power of this new design tool, consider a representative control channel having a total failure rate λ of 10^{-4} each hour. Typically, the relative probability of a hardover failure in control channels ranges between 0.1 and 0.9; let us assume 0.5. Therefore, the total failure rate of the channel consists of the sum of the active failure rate λ_a of 5×10^{-5} each hour and the passive failure rate λ_p of 5×10^{-5}. Assuming our system has two parallel channels, an active failure in one channel prevents the normal operation of the other so that the system fails. Therefore, the probability of a system failure Q is the sum of these failures. Passive failures allow the other channel to continue working. Q is the product of these failures because both channels must fail in order to constitute a system failure. The system failure is then the sum of the two terms

$Q = 2\lambda_a t + (\lambda_p t)^2$ where t is time and

$Q = 2 \times 5 \times 10^{-5} \times 1 + (5 \times 10^{-5} \times 1)^2$ where t = 1 hour

$Q = 10^{-4} + 2.5 \times 10^{-9}$

Note that the active failure term dominates by five orders of magnitude over the passive failure term. If all failures were passive, then

$Q = (10^{-4} \times 1)^2 = 10^{-8}$

The state of the art in fail-passive design can reduce the relative probability of active failures to about 0.1 percent. For our example, then $\lambda_a = 10^{-7}$ each hour, $\lambda_p = 10^{-4} - 10^{-7} \approx 10^{-4}$ each hour, and

$$Q = 2 \times 10^{-7} + (10^{-4} \times 1)^2$$

or

$$Q = 2 \times 10^{-7} + 10^{-8} = 2.1 \times 10^{-7} \text{ for 1 hour.}$$

This represents three orders of magnitude improvement in system reliability through fail-passive design techniques.

We can further compare the fail-passive approach to the conventional triplex voted method which requires three channels plus three comparators and voting logic. The failure rate λ_m of the monitor equipment per channel will typically be no more than 10 percent of the channel failure rate or 10^{-5} in this case. We can further make the blithe assumption that monitors are fail-safe (i.e., indicate their own failures) so that monitor failures do not cause system failure unless all three fail. In a conventional triplex system, the monitor votes out the first failed channel leaving two working ones. On a second failure the monitor cannot determine which channel has failed so both are turned off. Therefore, the system requires that two of the three channels work for success, and because the monitors do not differentiate between types of failures

$$Q = 3(\lambda t)^2$$

For our example

$$Q = 3\ (10^{-4} \times 1)^2 = 3 \times 10^{-8}$$

which is very close to the Q of the fail-passive system. However, the total failure rate of the triplex voted and fail-passive systems are 3.3×10^{-4} and 2×10^{-4} respectively. The relative cost, size, and weight will be in approximately this same ratio of 1.65:1.

SECTION IV

DISCUSSION OF FLY-BY-WIRE CONTROL

1. INTRODUCTION

The preceding section has described briefly the evolution of flight control systems, discussed the problems of mechanical control systems, and shown how flight control systems are evolving toward fly-by-wire. The benefits of fly-by-wire with the help of some examples will next be detailed, and then a fly-by-wire control system with some indications of needed development will be described functionally.

2. BENEFITS OF FLY-BY-WIRE

Our definition states that a fly-by-wire control system is an electrical primary flight control system employing feedback such that vehicle motion is the controlled parameter. Fly-by-wire provides a redundant integrated flight control system with the reliability and flexibility necessary to solve the increasingly complex problems of flight controls. This solution to the flight controls problem results in additional benefits. A reduction of control system weight of 150 to 750 pounds may be realized by fly-by-wire controls with recovery of a major part of the volume normally allowed for control linkages, cable motions, and artificial feel devices. The weight and space savings could be used for other aircraft subsystems or to improve aircraft performance. The weight reduction would especially enhance VTOL performance where the power to weight ratio is often marginal. Control system performance is improved by the elimination of the compliance, friction, and inertia of the mechanical linkages. Since the control system and airframe are mechanically uncoupled except at the actuator, the effects of airframe flexibility and temperature variations are reduced to a minimum. Fly-by-wire techniques provide these performance gains and weight and space savings with a reduced initial design effort and with simplified installation and maintenance. Fly-by-wire also provides a means of standardizing flight controls between aircraft and increasing the flexibility of cockpit installations. An additional benefit of fly-by-wire controls is their reduced vulnerability to battle damage. This fact alone makes fly-by-wire controls attractive to military users. To meet the control system reliability requirements, fly-by-wire controls employ redundancy techniques. Redundancy implies an increase in the number of components and an increase, therefore, in the number of maintenance actions required. However, modular packaging and failure reporting circuits lower maintenance time so that the overall maintenance costs will actually be less. The main disadvantage fly-by-wire has to overcome is the lack of confidence in system integrity caused by a distrust of nonmechanical system reliability.

A simplified diagram of the F-111 mechanical pitch/roll control was shown in figure 9 of Section II. Figure 26 shows an equivalent fly-by-wire system mechanization of this system. The relative simplicity of the fly-by-wire system is obvious even though it is highly redundant. Section VIII contains a comparative analysis of the weight, volume, and cost of the F-111 control system and a fly-by-wire equivalent. The results are shown in table XV of that section. The results show about a 50-percent reduction in weight and

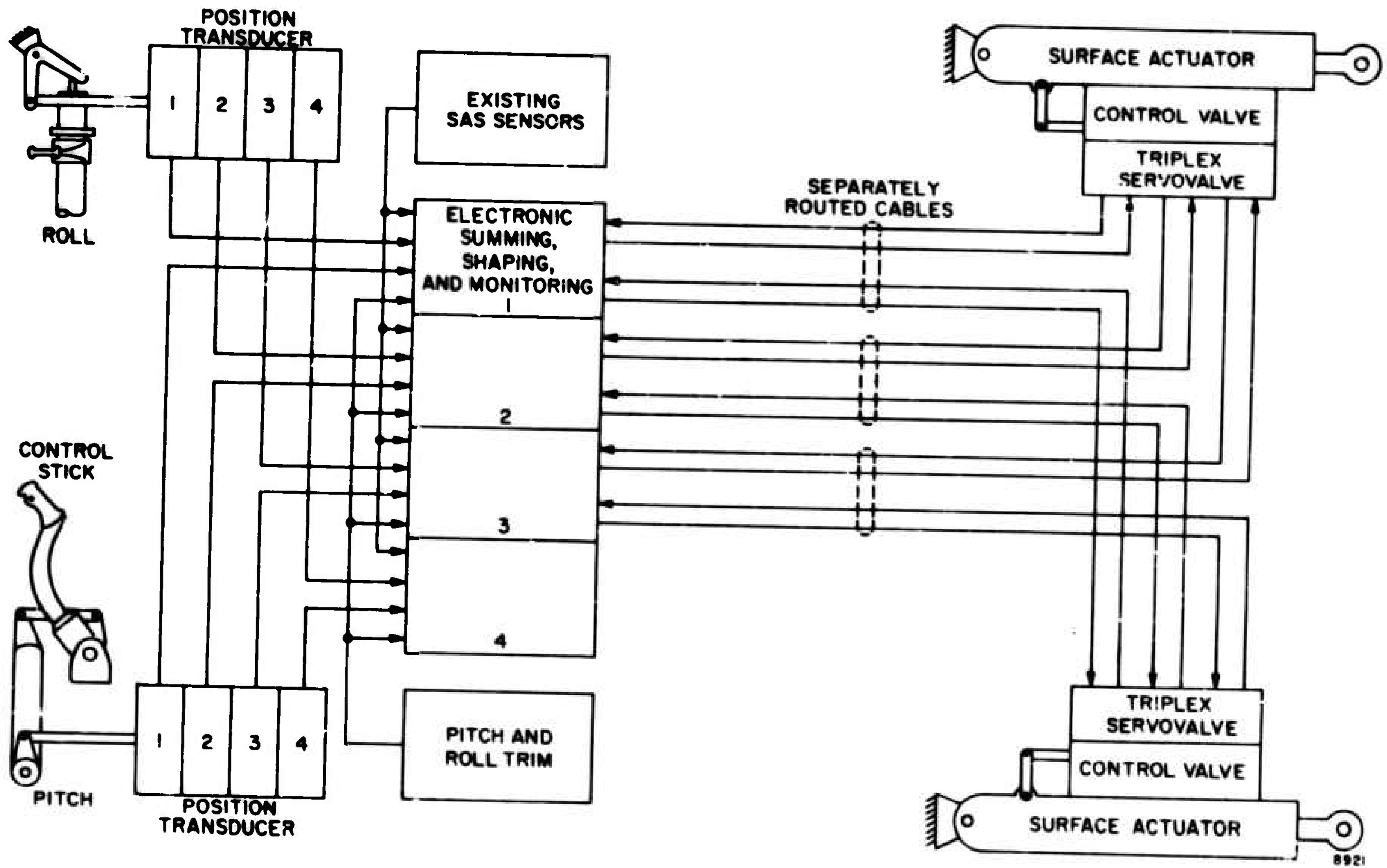

Figure 26
Equivalent Fly-By-Wire System

volume. The results of a similar weight analysis by General Dynamics are included for comparison. The General Dynamics weights essentially corroborate Sperry's estimates. A cost savings greater than the 15 percent shown would result because savings in design costs are not reflected. The F-111 will be discussed later.

On the XB-70 flight control system, North American originally proposed a fly-by-wire approach. Unfortunately, this approach was discarded in favor of a more conservative approach, and the design wound up mechanical. The schematic diagrams of the pitch and roll axes are shown in figures 27 and 28. The appendix contains a complete mechanical diagram of the system. It demonstrates the magnitude of complexity to be expected in such large aircraft as the SST, AMSA, and C-5A. At the conclusion of the design of the XB-70, the designers compared notes on the two approaches. They determined that the fly-by-wire approach would have saved 675 pounds and 90,000 of the 100,000 hours of design time.

Fly-by-wire would have provided many other advantages in the XB-70 according to the designers. In addition to eliminating the control linkage routing and fuel cell seal problems described in Section III, it would have also

a. Saved space
b. Provided better control resolution
c. Reduced inertia
d. Eliminated high-temperature bearings
e. Eliminated mass unbalance in the control system
f. Been more flexible to design changes
g. Been easier to make redundant
h. Reduced the interface problem with the other aircraft subsystems

Fly-by-wire would also allow the use of sidestick controllers which would allow moving the displays closer to the pilot and would reduce pilot-inertial coupling.

The CH-46 study, which is also detailed in Section VIII, demonstrates that fly-by-wire can cut the weight of the control system from 550 to 135 pounds. By similarity, the same fly-by-wire system would work in the CH-47 as well, and it would cut that aircraft's control system weight from 880 to 135 pounds. Furthermore, the maintenance time for each flight hour would be reduced by approximately two orders of magnitude. A Douglas study of the proposed DC-10 flight control system indicated that fly-by-wire would provide a weight savings of 230 pounds for a linkage system that is almost entirely cables. The weight savings for the F-111 and B-52H (Section VIII) would be 277 and 415 pounds, respectively.

The use of fly-by-wire techniques accrues some additional benefits because of its design flexibility and simplified interfaces. The Air Force has a study program, being performed by the Bendix Corporation, to determine whether the flight control system can be employed to alleviate gust loads on the airframe. The study, being performed on the B-52H, is trying to find out whether reducing the peak airframe stress levels can lengthen the service life of large expensive aircraft. The approach being used is essentially to add a CAS which modifies the control signals in such a way as to prevent

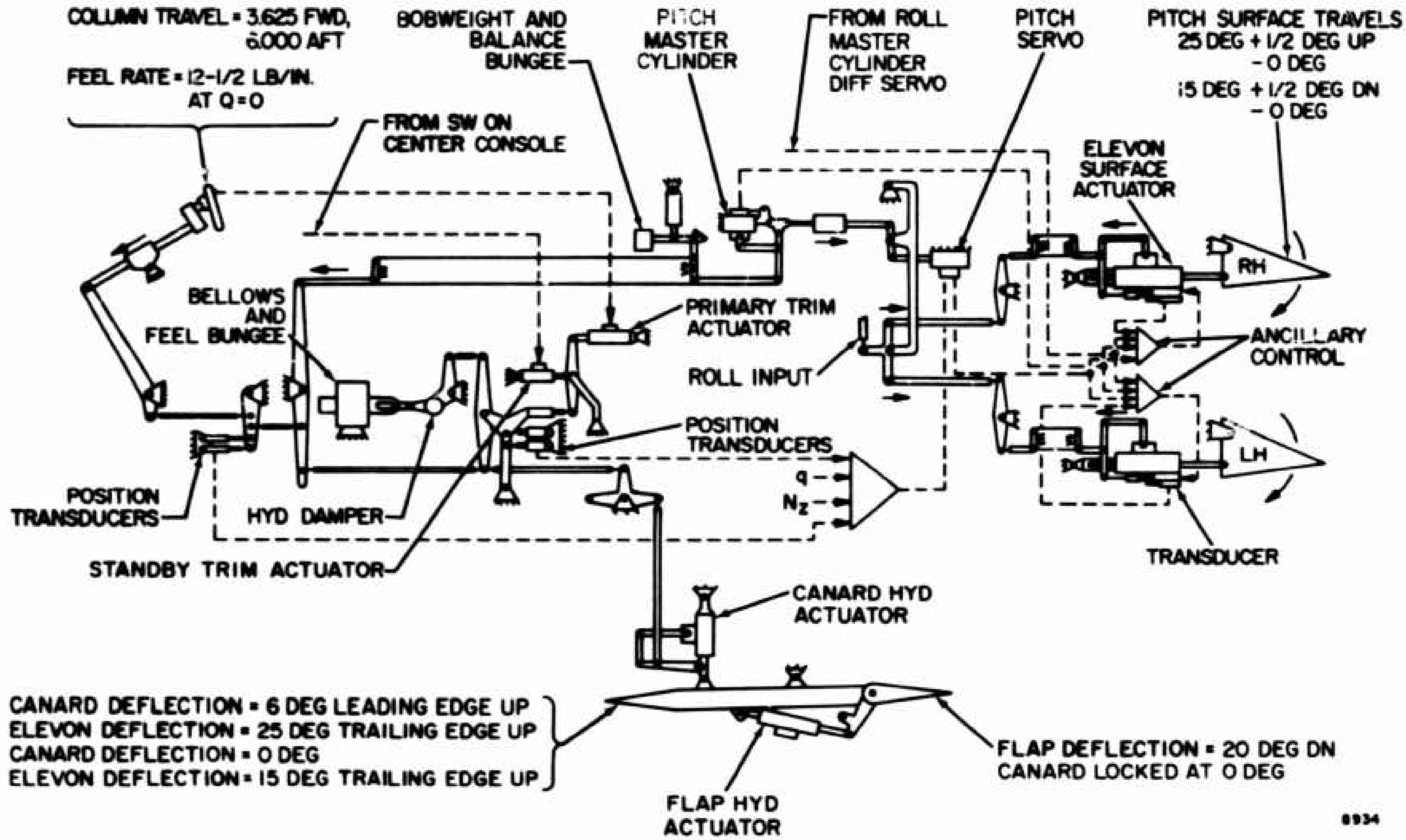

Figure 27
XB-70 Pitch Axis Control System

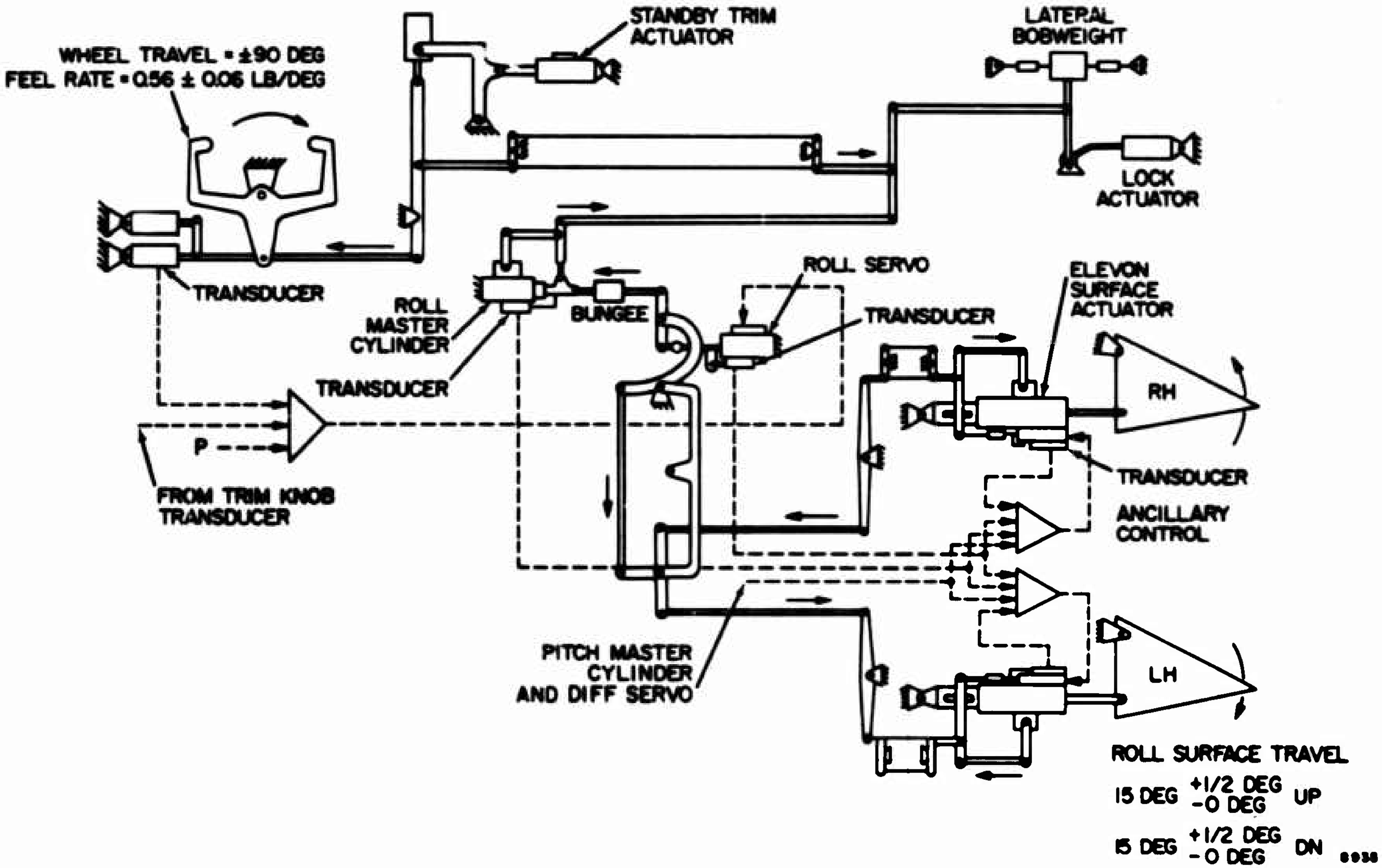

Figure 28
XB-70 Roll Axis Control System

large commands from overstressing the airframe and to attenuate body bending modes. Control signal modification may be done with precisely fixed filters or through adaptively-controlled variable filters. Since the means to add such a capability to a fly-by-wire system would already exist, an implementation could be readily incorporated. A similar situation exists with direct lift control in which wing lift is controlled directly (through the use of high-speed flaps, collective ailerons or spoilers, or boundary layer control) to improve flight path control particularly during landing. Such a scheme could be more easily implemented in a fly-by-wire system than a mechanical system. In fact, the Navy is adapting the A-7A to direct lift control through fly-by-wire techniques.

3. FLY-BY-WIRE SYSTEMS DESCRIPTION

From a systems viewpoint, the flight control system should allow the pilot to maintain direct and effective path control under all flight conditions with a minimum of error and effort. Direct and effective implies that simple commands should produce flight path corrections with adequate speed and precision. Effort includes mental as well as physical. The fundamental decisions on the control philosophy must be guided by intuitive principles, involving simplicity, reliability, and the integration of sensors, computations, and controls. A well-behaved system is best obtained through a closed-loop approach in which the desired flight path is the input and the actual flight path and dynamic response parameters to be controlled are fed back.

The pitch, roll, and yaw axes of the fly-by-wire control system that has evolved from our studies are shown in figures 29, 30, and 31. The technique is very similar to the command or control augmentation schemes employed in the F-111 and A-7A. The pitch axis employs the C* blend of pitch rate and normal acceleration feedback. The roll axis feeds back roll rate p, while the yaw axis feeds back yaw rate r, and lateral acceleration n_y. The requirements for n_y feedback depends on the airplane; it is generally required to eliminate sideslip during maneuvers thereby providing automatic coordination.

The closed-loop control systems in conjunction with the spring restrained control stick provide the necessary artificial feel. The feedback signals are compared with the command signals from the stick position (or force) transducers which have been shaped by a command model. The difference is an error signal which drives the control surface through a high gain servo. In operation, the higher the forward gain K, the less effect the aircraft dynamics have on the feel and the more the aircraft feels like the model. Aircraft feel, therefore, can be readily tailored for any aircraft (to personal preference if desired) using essentially the same system. Adaptive control can be added to optimize the response for all flight conditions by keeping the K as high as stability will allow. The more simple fixed gain (or manually varied gain) normally supplies adequate performance by selecting the highest gain usable at the worst case flight condition.

When the highest possible loop gains are used, the control system response approaches that obtained by integral control. Integral control provides all of the advantages of attitude displacement feedback for accurate path control while retaining good command response characteristics. Integral

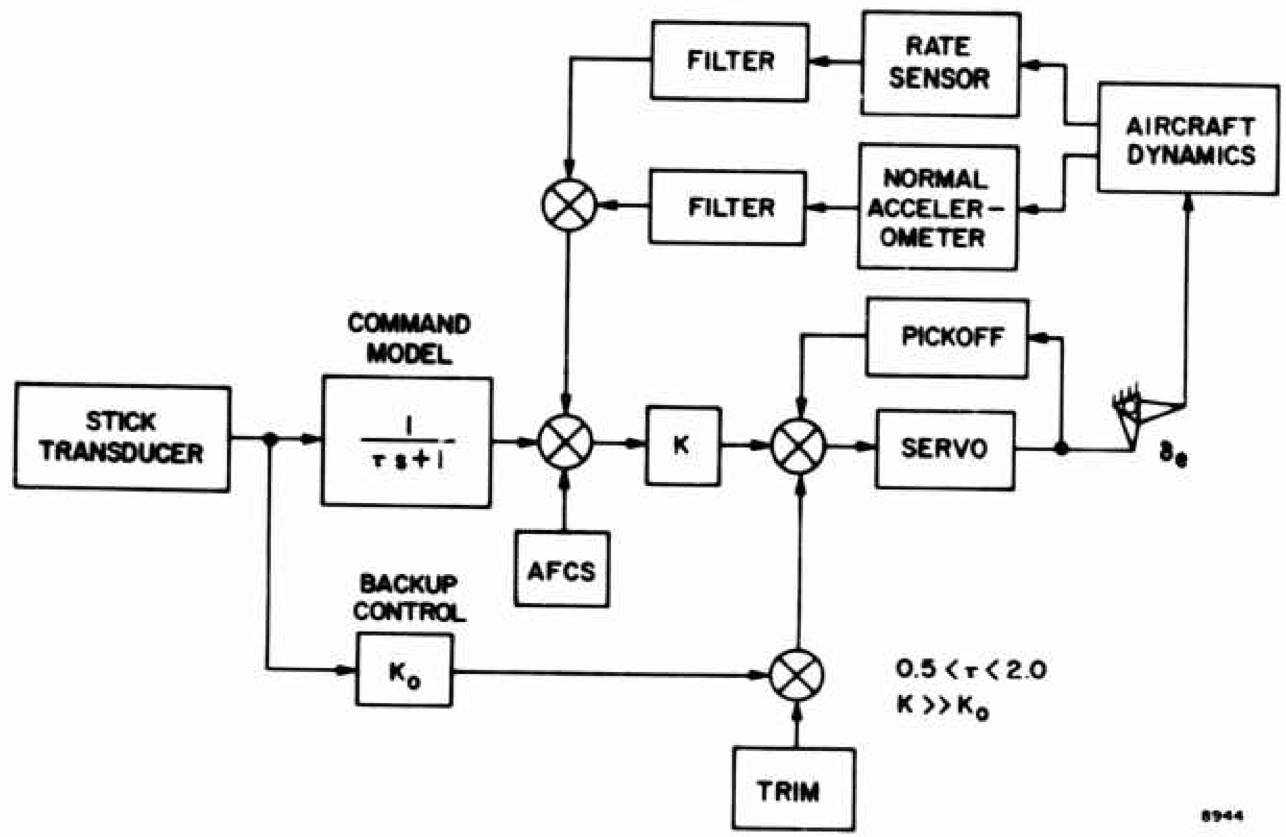

Figure 29
Simplified Block Diagram of Fly-By-Wire Control System - Pitch Axis

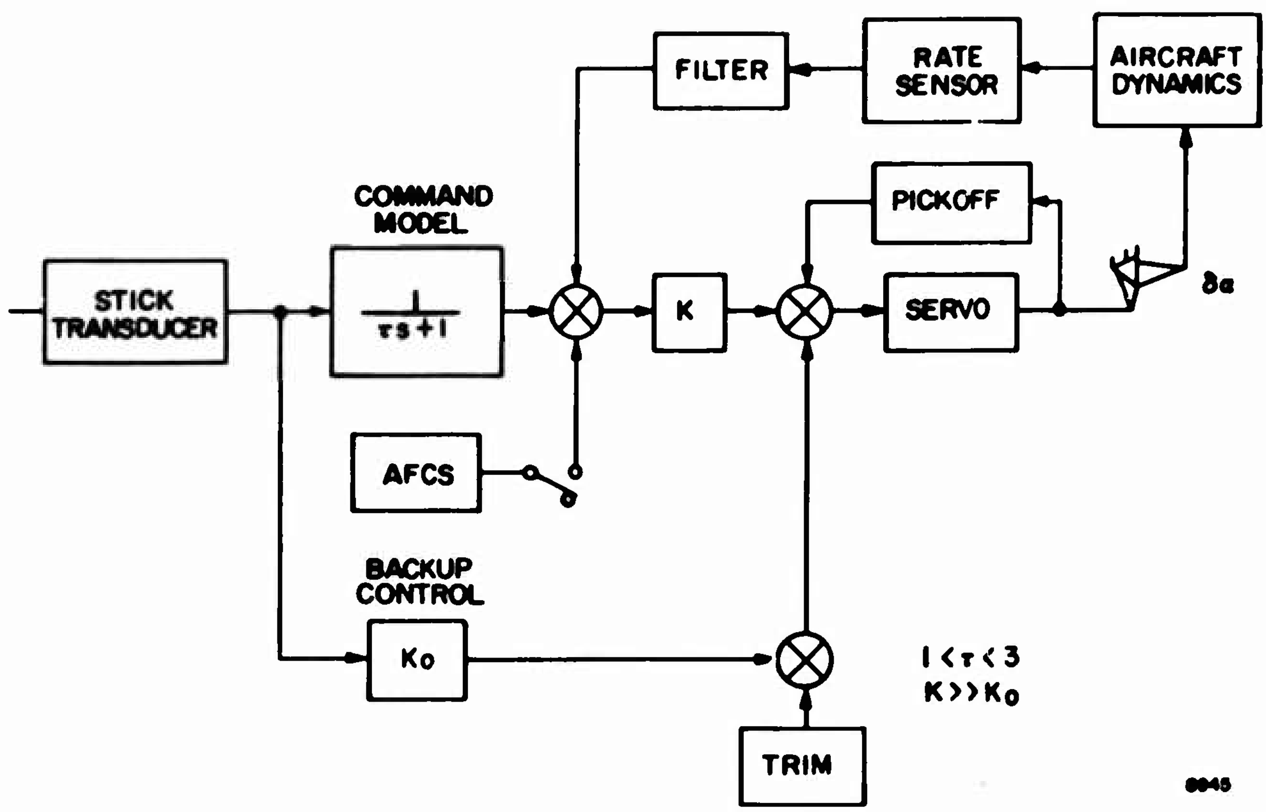

Figure 30
Simplified Block Diagram of Fly-By-Wire Control System - Roll Axis

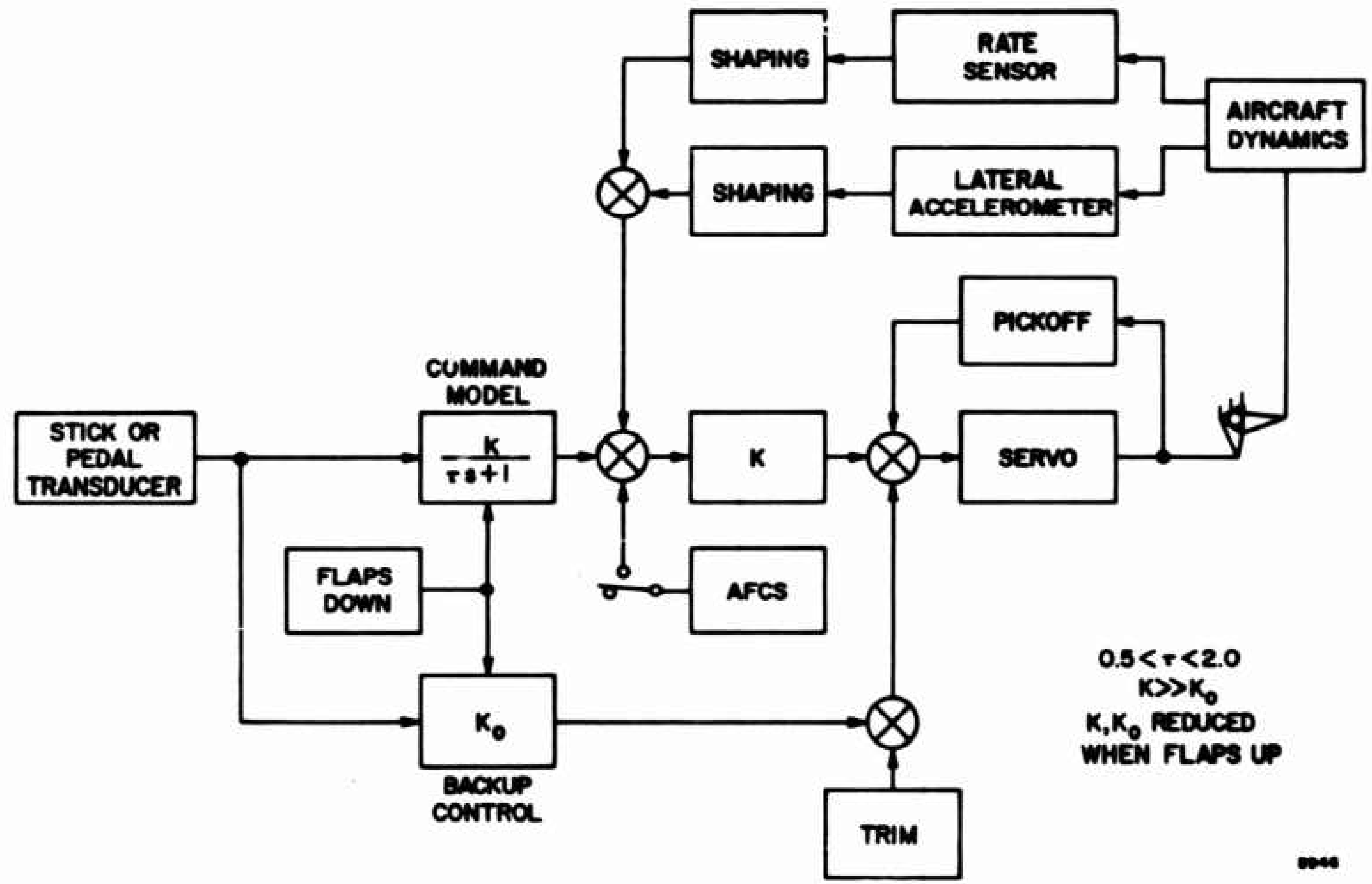

Figure 31
Simplified Block Diagram of Fly-By-Wire Control System - Yaw Axis

control may be provided by adding a single-degree-of-freedom rate integrating gyro (RIG) in the command path or by use of an integretor in the C* loop. This approach provides all-attitude capability while remaining independent of inertial platforms or attitude gyros. The pilot command through the RIG yields a proportional C* response. The command torques the RIG gimbal. The RIG integrates the aircraft rate response resulting in a gyroscopic torque on the gimbal thereby opposing the command torque. These two torques are equal at the commanded aircraft rate. Gimbal rotation is always very small so that the gyro can be said to be synchronized to the aircraft at all times. When the pilot removes the command, the RIG holds the aircraft at the existing attitude. Hence, there is all-attitude path control. Without command inputs the RIG acts as an attitude gyro by generating error signals proportional to changes in attitude. The RIG actually does this by integrating body rate, but the integral of rate is attitude. Only two RIG are required; one for the longitudinal axis, and one for the lateral axis. The lateral axis RIG is located in the roll command path, but it actually senses yaw rate. This takes advantage of the normal interaction between the roll and yaw axes to simplify the control system.

A comparison of the performance of the C* command system and integral control of C* would show that the latter would act like a neutrally stable airplane. That is, the aircraft would tend to stay in whatever attitude the pilot places it. The adaptive-gain C* and fixed (lower) gain C* systems would act less and less like a neutrally stable airplane. Integral control provides excellent path control which would be very beneficial to tracking tasks. The C* command system would not hold an attitude quite as well which would make it more comparable to present-day systems. Integral control has what some critics term a serious handicap. When approaching a stall condition, it will continue to maintain a fixed attitude. This tends to wash out the mild buffeting that precedes stall onset which would ordinarily alert the pilot. The system could continue to hold attitude on into the dangerous deep stall region. If stall warning devices of some type were not available to forewarn the pilot, he could find himself in deep trouble without realizing it. The C* command system would not have this problem because the pilot would notice the buffeting in time to recover. A neutrally stable airplane may be objectionable to pilots during combat maneuvers because of the complete loss of speed feel. C* command acts like an imperfect integral controller that retains a suggestion of speed feel particularly in the fixed-gain system. This is a subjective argument which bears verification. A decided advantage of the C* command system is that it does not require the RIG which is a relatively expensive and unreliable mechanism. Therefore, because of its relative simplicity and apparently more natural feel characteristics, the C* command should be used for the basic fly-by-wire system. Integral control should be added for long-term attitude control in the form of a slow integrator in parallel with the command system much like a series automatic trim function. Fast integral control for tracking and fire control can be added as required.

Artificial feel implemented by the C* command approach has several advantages over other methods.

a. It provides nearly neutral speed stability which permits tracking during rapid speed changes without trim.

b. Aircraft response conforms to angular rate at low speeds and normal acceleration at high speeds.

c. System is independent of airspeed or altitude.

d. It provides good command response while maintaining high gust damping.

e. It is flexible so that signals from other subsystems can be added.

f. Feel is independent of the aircraft; therefore, all aircraft of a class (e.g., fighters) could have the same feel and use the same components.

g. It utilizes sensors which nearly always already exist in the aircraft to augment stability. Therefore, additional components such as q-springs and bobweights are unnecessary.

In the system block diagrams, the direct (backup) path parallels the normal operational path. Because of the high gain of the normal control and the relatively low gain of the backup control, the normal control dominates until it is disengaged because of failures. Presence of the backup control creates a small bias which does not affect normal operation. The direct control path supplies a simple but sluggish backup control for the system with essentially no feel provisions. It will provide emergency control of the aircraft to get the pilot back to his base. Trim is applied either in series at the servo by an electrical bias in parallel, by an actuator on the control stick to position the neutral point, and/or through a separate trim actuator (such as to move trim tabs or the horizontal stabilizer). The pilot's controller is a spring-centered stick with an electrical output from either position or force transducers. The stick (or wheel) may be either a conventional type which is center located or a small stick which is located at one side. Minimum system redundancy will be either three parallel real channels and a model (simulated) channel or four parallel real channels. The reasoning behind this redundancy level is discussed in Section VI under Tradeoffs.

The general advantages of fly-by-wire control over the conventional mechanical designs are summarized as follows:

a. Improves control performance through better dynamic response control and the elimination of friction, backlash, hysteresis, compliance, inertia

b. Smaller installed weight and volume

c. Reduces total cost of ownership including initial, maintenance, and logistics costs

d. Better maintainability and logistics because of the reduction in the number of critical parts, easier access, and higher level of interchangeability between aircraft

e. Reduces vulnerability to minor structural damage, maintenance errors, and battle damage

f. Greater cockpit installation and orientation flexibility

g. Eliminates coupling into body bending modes

h. Reduces the required design effort

i. More flexible to design or performance changes

4. DEVELOPMENT OF FLY-BY-WIRE

The preceding discussion has shown where fly-by-wire technology stands today. Our present studies have shown that a practical fly-by-wire system is within the present state of the art. However, a great deal of reluctance to use it exists because of the lack of confidence in maintaining system integrity and because of a lack of familiarity of design groups with the available components and design techniques. The first logical step, therefore, is to build an experimental laboratory model of a fly-by-wire system to demonstrate its operation under simulated failures. This model will also demonstrate the use of state-of-the-art components and the effectiveness of existing design techniques.

Construction and evaluation of a laboratory experimental model of a representative fly-by-wire system will accomplish a number of ends. First, it will demonstrate the systems operation and performance under various failure conditions; second, it will provide data to establish performance requirements; third, it will provide a test bed for testing other techniques which may become available during the course of the development; and fourth, it will provide the data needed to design future flightworthy systems.

The next logical step is to convert an existing aircraft to fly-by-wire using the data from the experimental model. Then, by putting as many flight hours on it as possible, the integrity and practicality of fly-by-wire can be demonstrated. Successfully completing this step should provide the impetus to all those people in industry who are waiting for in-flight proof of fly-by-wire maturity.

We are interested in the capability of designing fly-by-wire systems today. Hence, we want to use as much available and proven hardware as possible. As shown in the following sections, very little development is required except in the actuator area. Here the prime concerns are the proper application of redundancy and the ability to prevent failures from adversely affecting system performance.

SECTION V

DESIGN CONSIDERATIONS AND SYSTEM COMPONENTS

1. DESIGN CONSIDERATIONS

A wide selection of components is available today for use in fly-by-wire systems. While little doubt of this fact exists in the industry, few people are aware of the choices and the optimum combinations in terms of weight, volume, cost, reliability, maintainability performance, and power drain. To determine which types of components are required, a number of general approaches to mechanizing fly-by-wire along with related system considerations will be discussed first. Component types will then be described.

One of the first factors to consider in establishing the approach to mechanizing a fly-by-wire system is the method of signal transmission. For the immediate discussion, the term "fly-by-wire" can be considered a catchall phrase meaning any nonmechanical technique for signal transmission. We could just as well fly by fluid or light or radio. We have already defined fly-by-wire as an electrical primary flight control system in which vehicle motion is the controlled parameter. Fly-by-fluid is similarly defined as the technique of fluidically transmitting all control signals from the pilot's station to the control surface actuators. This technique has the potential advantage of not requiring electrical power. Such an approach was at one time considered for the Concorde SST. In a fly-by-fluid system, control stick motion modulates a pressure control valve so that it operates as a position transducer, that is, the output pressure is proportional to stick position. While such a transducer does not presently exist, little development would be required in designing one. The pressure change is transmitted along a pair of hydraulic lines to operate the servo control valve. Mechanical feedback would be used on the actuator. Several fluidic SAS are being developed which could be summed hydraulically or an electrical SAS could be summed through an electrohydraulic valve. Actuator monitoring would be hydraulic. While such a scheme appears to be workable, it has several basic drawbacks. Because of the compressibility of hydraulic fluid, a signal time delay results. The delay time or lag in a line 100 feet long is about 24 milliseconds based on a typical speed of sound in the fluid of 4,200 feet each second. Because of line reflections, the system could resonate at about 6.6 hertz. To maintain a resonable phase margin, the frequency response would have to be limited to about 4 hertz. The acceptability of the actual limit, and hence the fly-by-fluid technique, depends on the application. On large aircraft, it may well be unacceptable because of the very long lines required. Other drawbacks include the weight of the pressure control lines needed for a redundant system which would very likely be triplex. These lines are in addition to the actuator supply lines. The trim input transducer may require development. It would provide a control pressure bias. An acceptable artificial feel mechanization would also require development. In summary, the fly-by-wire fluid technique would have the advantage of not requiring electrical power, but it has the disadvantages of requiring additional development and having limited frequency response.

A fly-by-light technique may also be considered briefly. Currently available components require the use of electrical transducers with signal conversion to light for transmission along optical fiber bundles. Reconversion to an electrical signal is required before commanding conventional electrohydraulic servos. The advantages of optical transmission are that stray signal pickup and interference are eliminated, interchannel isolation is ensured because common paths or short circuits cannot occur, and a very high degree of redundancy is available within the glass fiber bundles. For example, a 1/8-inch bundle contains about 70,000 fibers, each 0.0005 inch in diameter. These very small fibers are very flexible and strong. Fiber bundles are currently used in some aircraft for observing in flight (from the cabin) the raising and lowering of the landing gear or the operation of fuel cells and control surface actuators. The disadvantages include the relatively high cost of fiber bundles of about a dollar for each foot and the very high optical attenuator along the bundles. The attenuation is approximately 50 percent in 6 feet. In this case, pulse-modulated signals would be desirable. As will be shown later, pulse or binary signals of any type are undesirable. Since the high signal attenuation is also undesirable, optical transmission can be ruled out.

The fly-by-radio technique transmits control signals via v-f energy thus eliminating wires altogether. However, this technique can be ruled out immediately because of the possibility of interference with the ship's instruments and radio and radar equipment and because the many bulkheads within the airframe effectively prevent signal continuity.

The conclusion of this discussion is that electrical signal transmission is the desired technique within the present state of the art in control system design. The next important factor to consider is the signal format or type since this grossly affects or is affected by the available equipment and its complexity. The signal type can be divided into five categories.

a. DC
b. AC
c. Pulse modulation
d. Digital
e. Multiplex

DC and suppressed carrier ac are the only commonly used signal forms in aircraft control systems. All available servovalves require dc drive current. Further, signal shaping and interchannel summing are best done with dc. Shaping with ac signals requires a supply frequency that is much more stable than is available in aircraft supplies. For a similar reason, summing signals from channels powered by different supplies is difficult if not impossible because the supplies are difficult to hold in phase let alone to hold at the same frequency. AC amplifiers are commonly used to eliminate the effects of amplifier drift, but amplifiers are now being built having nearly the drift characteristics of an analog computer amplifier. For example, drift can be readily maintained within ±0.25 percent of full scale output over the temperature range -131°F (-55°C) to +255.2°F (+125°C) with ±0.05 percent capability available. AC electronics are employed in fail-passive designs to eliminate the effects of hardover failures. Failures produce large dc outputs to which the ac circuits are insensitive. Fail-passive design was discussed for reliability reasons in Section III. Finally, since

inductive transducers are preferred, some ac electronics are required to process their outputs. Although other signal formats have advantages of noise immunity or circuit simplicity, their overall complexity plus the functional simplicity of the fly-by-wire system makes the use of other than simple dc or suppressed carrier ac difficult to justify. With the possible exception of digital signals, all other formats require complex conversion equipment at the transducers and other inputs and again at the servos. Such signal formats include frequency modulation, various forms of pulse modulation (e.g., rate, width, amplitude, code and delta), and forms of multiplexing (e.g., frequency and time). Furthermore, even simple signal shaping, such as a lead or lag function, is often difficult to perform without additional circuit complexity or reconversion to dc. Digital signal formats (whole word or incremental) could be applicable where digital transducers and acutators are available because no conversion equipment would be required. Signal shaping still requires excessive complexity for the whole word format but not for incremental computation. In either case, a practical digital servoactuator is not available within the current state of the art. Present digital actuator designs, which are being developed by a number of companies, tend to be large and complex. We conclude then that both dc and suppressed carrier ac should be used in fly-by-wire systems because system complexity and cost are minimized, and reliability is maximized. DC signals should be used for signal shaping, summing, and for the servovalve drive. AC signals should be used in conjunction with transducers and in fail-passive circuits.

To determine the components required for a typical fly-by-wire control system, we can refer to the block diagram shown in figure 32 which might result from a preliminary design. The components can be categorized into the control stick, transducers, transmission line, summing junctions, electronics, actuators, artificial feel sensors, and the trim actuator and switches. The hydraulic and electrical power supplies are excluded. Table I indicates the developmental status of the various types of transducers and sensors that are applicable to fly-by-wire systems. Table II summarizes the important transducer characteristics. Table III shows the performance requirements for the transducers and sensors.

2. CONTROL STICK

a. Introduction

A control stick may be a center stick or wheel or a small sidestick hand controller. The selection of one of these types lies basically in the realm of human factors and depends heavily on the desired cockpit organization. The following discussion describes sidestick controllers because they enhance the benefits of fly-by-wire systems more than center sticks or wheels and because they may very well not be as familiar to the reader. Sperry does not contend that center sticks or wheels should not be employed in fly-by-wire systems, but they are less desirable because of their larger size and weight and their relative position in the cockpit. The discussion considers the type of stick, articulation, cross coupling, trim control, and parameter adjustment. The design of a typical sidestick controller is also presented. The design is based on Sperry's extensive background and on the results of previous military development programs in this area. (Refer to references 19 through 25.)

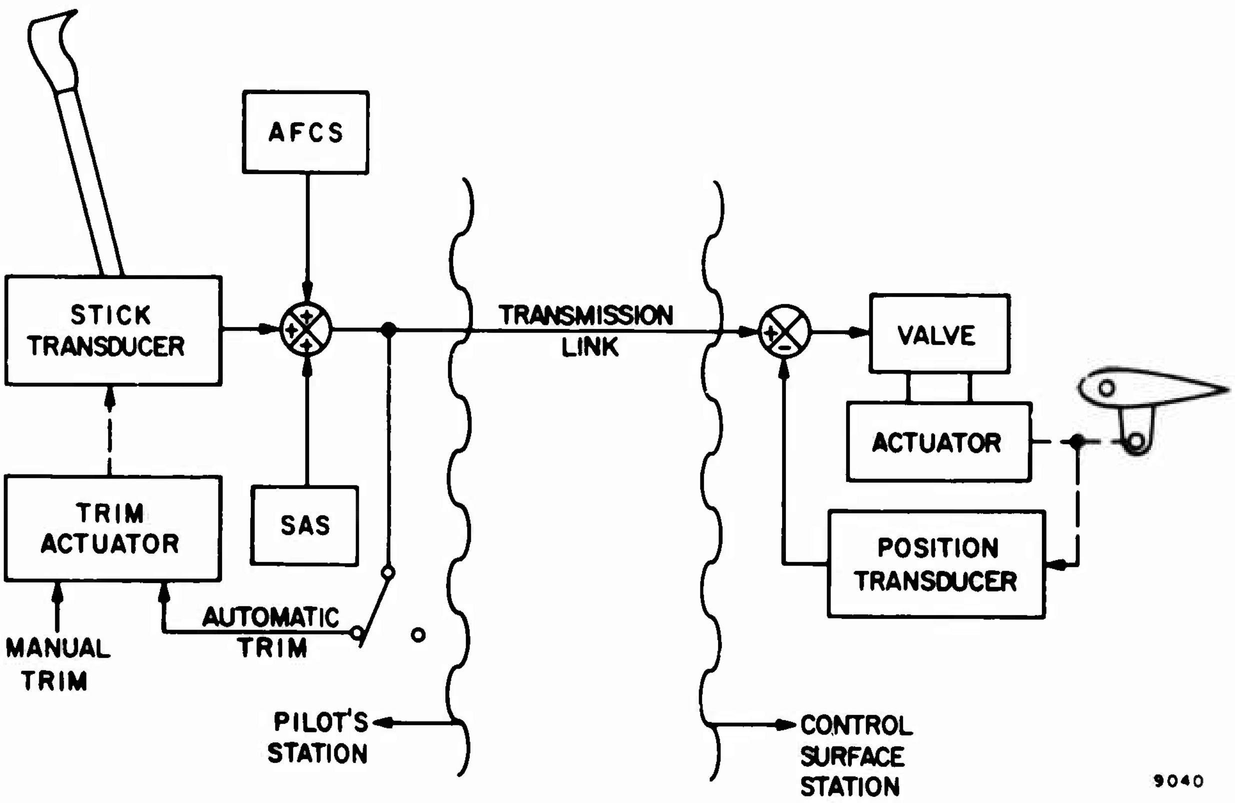

Figure 32
General Fly-By-Wire Configuration

TABLE I

SUMMARY OF TRANSDUCER AND SENSOR TYPES

Function	Type	Status	Development Required
Rate Sensing (Angular)	Single-degree-of-freedom, spring-restrained gyro	Highly developed, available from many established sources	Improve reliability, primarily bearing life
	Fluid Tube Gyro	Developed by Sperry to high performance	Develop for low cost, moderate performance
	Vibragyro	Numerous devices of this type have been investigated. Vibrating beam and cylindrical models of the internally driven types are worthy of further development	Extensive development is required, including analysis, fabrication, and testing of these devices. Low signal levels and null shift problems must be overcome
	Laser Gyro	Well along in laboratory stage of development by Sperry and other companies	Develop for low cost, moderate performance applications. Improve gas-tube laser life
Acceleration Sensing (Linear)	Force-Balance Torque-Restored Pendulum	Highly developed, available from several established sources	Improve reliability, develop for low cost applications
	Variable Reluctance Sensor	Highly developed, low cost	Improve accuracy, simplify and miniaturize structure
	Vibrating Wire Sensor	Highly developed for very accurate applications	Develop for low cost, moderate accuracy applications

TABLE I (cont)

SUMMARY OF TRANSDUCER AND SENSOR TYPES

Function	Type	Status	Development Required
Pressure Sensing (Air Data)	Vibrating Diaphragm	Well along in development, state-of-the-art performance, low cost	Develop production capability
	Strain gage	Highly developed, available from several established sources, costly	Improve high-level output semiconductor types to reach performance now available from low level output strain wire types. Develop for low cost, high performance applications
Differential Pressure Sensing (Hydraulic)	Variable Reluctance	Highly developed, available from several established sources	Develop low-cost design with integral solid-state electronics to minimize sensitivity to excitation voltage variations. Ruggedize
	Strain gage	Highly developed, available from several established sources	Improve high-level output semiconductor types to reach performance now available from low output strain wire types. Ruggedize
Position Sensing	Variable Resistance	Highly developed, reliable in film or plastic element types, low cost	Develop to minimize sensitivity to dust, hydraulic fluid contamination
	Linear Variable Differential transformer; Synchro	Highly developed, reliable, low cost	None

TABLE I (cont)

SUMMARY OF TRANSDUCER AND SENSOR TYPES

Function	Type	Status	Development Required
Position Sensing (cont)	Shaft position encoder	Highly developed, available from numerous sources	Develop for high reliability, low cost
Force Sensing	Variable Reluctance	Highly developed, available from several established sources	Develop for low cost with integral solid-state electronics to minimize sensitivity to excitation voltage variations
	Strain gage	Highly developed, available from several established sources	Improve high-level output semiconductor types to reach performance levels now available from low output strain wire types
	Vibrating wire	Well developed for other applications	Develop packaging and electronics for low cost, moderate accuracy force-sensing applications

TABLE II

SUMMARY OF COMPONENT CHARACTERISTI

Component	Signal Type	Power Level	Size	Weight	Cost (dollars)	λ Each 10^6 hr
Control Stick Position Transducer						
LVDT	ac	Low	1/4 D x 7/16 in. L to 7/8 D x 14 in. L	1 to 5 oz	50 (short) 100 to 300 (long)	12
		High	3 D x 4 in. L	15 lb		
Synchro	ac	Low	Size 8, 3/4 in. D	1 to 3 oz	40 to 100	8
Potentiometer	dc ac	Low		1 to 6 oz	5 to 25	100 to 200
Digital Encoder	binary code	Low	2 ± 1/4 in. D 1-1/4 to 3 in. L 8, 11 type	1/4 to 1 lb	200 to 600	100
Control Stick Force Transducer						
E-Pickoff on a Machined Spring	ac	Low	1-1/2 x 14 in. series control link 3 D x 11 in. link in column	2.5 lb 2.7 lb	800 to 1,000 for each axis	12
Fail or Wire Strain Gage on Load Beam	ac dc	Low	1 to 10 in.^{3} for load beam	1 to 2 lb for load beam	800 to 1,000 for each axis	24 for each bridge
Semiconductor Strain Gage on Load Beam	ac dc	Low	1 to 10 in.^{3} for load beam	1 to 2 lb for load beam	800 to 1,000 for each axis	24 for each bridge

TABLE II

SUMMARY OF COMPONENT CHARACTERISTICS

	Weight	Cost (dollars)	λ Each 10^6 hr	Tensile Strength psi x 10^{-3}	Comments
L	1 to 5 oz	50 (short) 100 to 300 (long)	12		
	15 lb				
D	1 to 3 oz	40 to 100	8		Includes induction potentiometer, linear transformer, linear synchro.
	1 to 6 oz	5 to 25	100 to 200		Conductive plastic has best life.
	1/4 to 1 lb	200 to 600	100		1 turn = 100 to 620 counts. Requires 10 KC to 200 KC interrogation signal plus electronics for readout.
.	2.5 lb 2.7 lb	800 to 1,000 for each axis	12		Experience shows current technology can produce more repeatable E-pickoff than strain gage type. Used in several systems.
	1 to 2 lb for load beam	800 to 1,000 for each axis	24 for each bridge		
	1 to 2 lb for load beam	800 to 1,000 for each axis	24 for each bridge		12 volts excessive, 3 volts output, 0.5 percent full-scale linearity, 0.2 percent full-scale null, maximum deflection 0.03 inch, 300 pounds overload capacity. Deposited type eliminates bonding.

TABLE II (cont)

SUMMARY OF COMPONENT CHARACTERISTICS

Component	Signal Type	Power Level	Size	Weight	Cost (dollars)	λ Each 10^6 hr	Te St psi
<u>Summing Junction</u>							
Transistor Amplifier	ac dc pulse	Low	TO-5 or 1/4 x 1/4 in. flatpack	Negligible	25 to 50	1	
			1 x 1 x 1/2 in.	1 oz	40 to 200	3 to 6	
Transformer	ac	Low	0.04 to 1 in.^{3}	0.1 to 1 oz	5 to 10	0.1 to 1	
		High		1 to 20 lbs	5 to 50	0.1 to 2	
Optoelectronic Transistor	ac dc pulse	Low	TO-5	Negligible	25 to 125	1	
Adder/Register	binary	Low	1 to 10 TO-5 or 1/4 x 1/4 in. flatpack	Negligible	60 to 100	0.1 to 1	
Fluidic Amplifier	dc binary	Low			40 to 200	1 to 6	
Resolver (Modified)	ac	Low	Size 8, 3/4 in. D	1.5 oz	40 to 100	8	
<u>Transmission Line</u>							
Electrical Wire	ac dc pulse	Low or High					
Stranded copper			#20	11 lb each 10^{-3} feet (shielded)	20 each 10^3 feet		38
Reinforced copper							50
Copper-clad steel				5 lb each 10^3 feet (unshielded)			100

TABLE II (cont)

SUMMARY OF COMPONENT CHARACTERISTICS

Weight	Cost (dollars)	λ Each 10^6 hr	Tensile Strength psi x 10^{-3}	Comments
Negligible	25 to 50	1		Microcircuit.
1 oz	40 to 200	3 to 6		Discrete components.
0.1 to 1 oz	5 to 10	0.1 to 1		
1 to 20 lbs	5 to 50	0.1 to 2		
Negligible	25 to 125	1		
Negligible	60 to 100	0.1 to 1		
	40 to 200	1 to 6		
.5 oz	40 to 100	8		Used with fail-passive servo only. Has two orthogonal stator windings and one rotor winding.
1 lb each 10^{-3} feet (shielded)	20 each 10^3 feet		38	
			50	Three strands stainless steel and 16 strands copper.
5 lb each 10^3 feet (unshielded)			100 to 150	Resistivity two and one-half times copper.

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TABLE II (cont)

SUMMARY OF COMPONENT CHARACTERIS

Component	Signal Type	Power Level	Size	Weight	Cost (dollars)	λ Each 10^6 hr
<u>Transmission Line (cont)</u>						
Optical Fiber Bundle	pulse	Low	0.03 to 0.4 in. diameter		<1/ft	
Hydraulic Line	dc pulse	Low or High	1/4 to 1/2 in. O.D.			
<u>Hydraulic Servovalve</u>						
Flapper Nozzle with Power Spool (flow control)	dc 8 to 100 ma	High	1-1/2 x 1-1/2 x 1-1/2 in. to 2 x 2 x 3 in.	0.4 lb 0.6 lb	300 to 500	250 to 500 λ active = 50 to 100
Jet-Pipe with Power Spool (flow control)	dc 8 to 100 ma	High	2 x 2 x 2 in.	0.6 lb	300 to 500	250 to 500 λ active = 25 to 75
Jet-Pipe, Single-Stage (pressure control)	dc 8 to 100 ma	Low	2 x 2 x 2 in.	0.5 lb	300 to 500	100 to 200 λ active = 0.3
Acceleration Switching	pulse 12 ma to 500 ma	High	2 x 2 x 3 in.	0.5 to 1.0 lb	200 to 400	N/A

TABLE II (cont)

SUMMARY OF COMPONENT CHARACTERISTICS

eight	Cost (dollars)	λ Each 10^6 hr	Tensile Strength psi x 10^{-3}	Comments
	<1/ft			Transmission loss 10 percent per foot. Fiber diameter 10 microns (0.0005 in.) to 75 microns (0.003 in.). 1/8 in. diameter bundle has about 70,000 fibers. Can be coherent.
4 lb 6 lb	300 to 500	250 to 500 λ active = 50 to 100		Standby leakage flow generally less than jet-pipe; has higher pressure gain.
6 lb	300 to 500	250 to 500 λ active = 25 to 75		Jet-pipe is much less susceptible to clogging than flapper nozzle.
5 lb	300 to 500	100 to 200 λ active = 0.3		Essentially no moving parts; jet-pipe acts like cantilever spring. Used in fail-passive actuator.
5 to 1.0	200 to 400	N/A		

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TABLE III

TRANSDUCER AND SENSOR REQUIREMENTS FOR FLY-BY-WIRE SYSTEMS

Function	Range	Threshold and Resolution	Accuracy (tolerance)	Null Stability	Remarks
Rate Sensing (Angular) Roll	±200 deg/sec	0.1 deg/sec			
Rate Sensing (Angular) Pitch, Yaw	±30 deg/sec	0.01 deg/sec	±1 to ±5 percent	0.15 deg/sec	
Acceleration (Linear	+7 to -3 g and ±1 g	7 x 10^{-5} g	±3 to ±10 percent	0.1 percent	
Pressure (Air Data)	1 in. Hg to 31 in. Hg ABS 0 to 15 in. Hg DIFF	0.001 in. Hg	±0.05 percent	Included in accuracy tolerance	Pressure transducer accuracies are specified for air data applications including cockpit instruments. Accuracy may be degraded by a factor of 100 for fly-by-wire applications.
Differential Pressure (Hydraulic)	3000 psig	3 psi	±1 to ±5 percent	30 psi	
Position (linear)	0 to ±0.3 in. 0 to ±1.0 in.	0.05 percent	±1 to ±5 percent	0.1 percent	
Position (Angular)	±20 deg	0.01 deg	±1 to ±5 percent	0.1 percent	
Stick Force Sensing	0 to ±35 lb 0 to ±100 lb	0.1 percent	±1 to ±5 percent	0.1 percent	

b. Moving Versus Rigid Stick

The possibility of using a pure force-sensing (rigid) stick is attractive in that problems with pivot location and articulation, discussed in subsequent paragraphs, are nonexistent. Development work by the military has shown that pilot performance with rigid stick controls is acceptable for many tasks; however, it is inferior to that with a moving stick particularly for high-demand tracking tasks.

c. Controller Size and Shape

Side controller evaluation programs have studied various stick shapes and sizes from conventional large center stick grips to low-force-gradient "pencil sticks" of 1/8-inch diameter. Flat handrest type controllers using the tracking ball principle have also been evaluated. Although accustomed to large grips, pilots frequently criticize their use in side control applications in which the rotational pivot is near the base of the grip; test results verify that more precise control is possible with the use of a stick of diameter more compatible with the limited motion of the side stick. Use of a pencil-type control is impractical not only because it would at present be a radical departure from the ordinary but also because the stick must provide for trim and interlock controls.

Another frequent objection to the use of conventional sticks as side controllers is the necessity of moving the hand upward from its normal position to reach the trim controls. A grip having a moderate diameter and height would meet the requirement for precise control and provide adequate volume for incorporating the necessary switches. A detailed component description of a practical stick grip is provided in paragraph V.2.g.

d. Articulation

The selection of pivot location for a side controller has been critical to the success of previous designs. The simplest mechanization is to pivot both pitch and roll (and yaw in a three-axis controller) below and near the base of the stick grip. This design requires translational rather than rotational hand motion, this has been considered disadvantageous particularly in a high-g environment in which the forearm is by necessity restrained. In three-axis controllers of this simple design, yaw-roll cross coupling is a particular problem which can be only partially overcome by the use of high detent forces or by providing free arm motion.

Various gimbaling techniques have been developed to overcome these problems. Wrist-pivot controllers in which roll and/or pitch motions are gimbaled back to the wrist have been studied. Pitch rotation about the grip center and roll rotation about the axis of the forearm as well as double-link (articulated) sticks have also been studied. Additional difficulties have arisen with each complexity; mass balancing becomes more complex, friction and lost motion increase, use of new muscles is required, wristlock or awkward control is observed. The use of wrist-pivot longitudinal cyclic control, for example, results in an up-down hand motion effect rather than fore-aft, requiring some retraining on the part of the pilot.

Much benefit has been derived from these programs; however, they indicate that as long as controller motion is restricted to certain limits the simple pivot technique is best for applications such as the one in question. Even the Gemini rotational side controller has evolved to a relatively simple design. Therefore, a short-throw, simply pivoted controller that is integral with the armrest and uses a broad handrest at the base of the grip should be used. With the elbow on the armrest and the heel of the hand resting on the stick, the rotational and slight translational hand motions will be natural and analogous to current practice of resting the arm on the right leg while using the center stick.

The controller should be provided with dual force gradients in both roll and pitch to meet the feel requirements as discussed in Section III. Control motions within the range of normal operation have a relatively low force gradient; beyond this range, soft stops engage and a higher force gradient is provided to the limit stops at the maximum stick throw.

e. Trim Control

The side controller normally includes a short-throw, four-way roll/pitch beep trim switch at the top of the stick. Location of the trim control has not proved very satisfactory in several previous side controllers because the pilot had to change hand position to actuate the control. The hand should not have to move up the stick to operate the trim switch; switch operation should not cause inadvertent stick motions, and vice versa.

The question of beep versus wheel trim always arises, and no single design has proved universally satisfactory. The current tendency particularly in high-performance, fixed-wing aircraft is toward wheel (displacement) trim (X-15, F-8). For helicopter/VTOL application, however, beep trim is more satisfactory.

In present-day aircraft, retrimming is accomplished as follows. As a result of a change in flight condition, the pilot finds himself holding a constant force on the stick, displaced from trim reference position. He actuates the trim switch, driving the magnetic brake/trim actuator to shift the zero-force trim reference point toward the new stick position. His stick force reduces at a constant rate. When he feels the force entirely relieved, he stops trimming, and a new trim reference position is therefore established. This technique results in smooth readjustment of trim without attitude oscillation.

Using a side controller without mechanical trim feedback to the stick, the pilot has to return the stick position to center while retrimming. With a wheel-type trim control, the pilot probably will have difficulty adjusting the trim wheel continuously and smoothly and moving the stick at the same time. Attitude oscillations will probably result. With beep trim, precise control of the trim switch is not required; actuation of the trim switch requires only that the pilot move the stick at a constant rate toward neutral. Since this rate is the same at all flight conditions, the pilot should readily learn how to retrim without causing oscillations. This technique is more closely related to present practice than would be the use of a wheel trim control.

A block diagram in figure 33 shows how the trim system functions. For simplicity, the figure shows only one redundant channel of one axis and excludes safety monitoring. Actuation of the momentary trim switch drives the integrator at a constant rate; the change in integrator output, up to the trim limit stops, is proportional to the time the switch is held down. Trim output then passes to the servo amplifier where it sums with the stick position signal. Trim output should also be displayed on a cockpit trim meter for pilot monitoring. Use of a cockpit trim switch, as shown, is suggested to provide two functions: automatic trim-to-neutral for preflight check list, and trim cutout in event of trim failure.

The trim system should be fail-operational and fail-safe. This feature is compatible with the fail-safe philosophy for SAS and related subsystems, and it is an improvement over the commonly nonredundant trim systems. Prevention of runaway trim by the fail-safe monitoring is often essential to the safety of the aircraft; although loss of trim is less critical.

f. Requirements for Adjustability

The side controls and armrests should provide adequate adjustments to accommodate the anthropometrical variations among pilots. Sperry has previously designed an elaborate study-program side controller which also provided simple in-flight adjustment of force gradient, detent force, and damping. From this and other programs generally acceptable specifications for these parameters have resulted. Therefore, a highly flexible design would be unnecessarily complex. Selection of stick characteristics and adjustment range can be based on results from these previous military programs; typical specifications are provided in the following sections.

g. Typical Sidestick

A typical fly-by-wire sidestick controller, shown in figure 34, provides quadruple output signals proportional to the displacement of the stick grip from the neutral position in both the longitudinal and lateral axes. In addition, a four-position trim switch, a pushbutton interlock switch, and a trigger switch are provided at the top of the stick grip. This sidestick controller includes an integral right-hand armrest for the pilot's seat.

The sensor design consists of a two-axis gimbal system with conventional coil springs for primary spring restraint. Two separate spring rates are provided in all axes of control. A fairly light spring rate is provided during the initial displacement of the grip and a stiffer spring rate is provided during the last segment of grip movement. The grip angle where the stiffer spring rate is contacted is ±10 degrees in the pitch axis and ±7 degrees in the roll axis.

The controller includes a padded elbow rest which is adjustable fore and aft. The elbow rest is lightly spring loaded to the full forward position and adjustment is made by depressing the adjustment pin, moving the rest rearward to the desired position, and releasing the adjustment pin. The rest will remain locked in this position until readjusted.

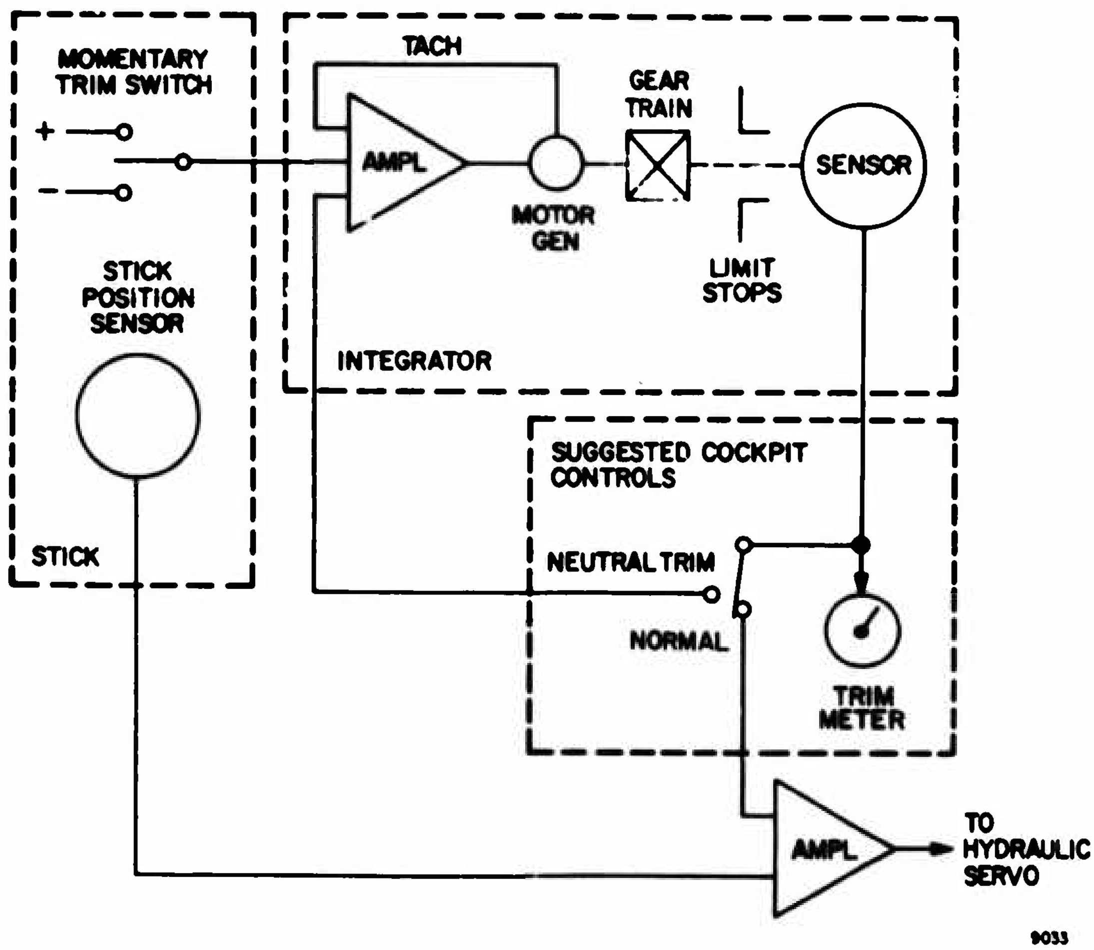


Figure 33
Trim Technique

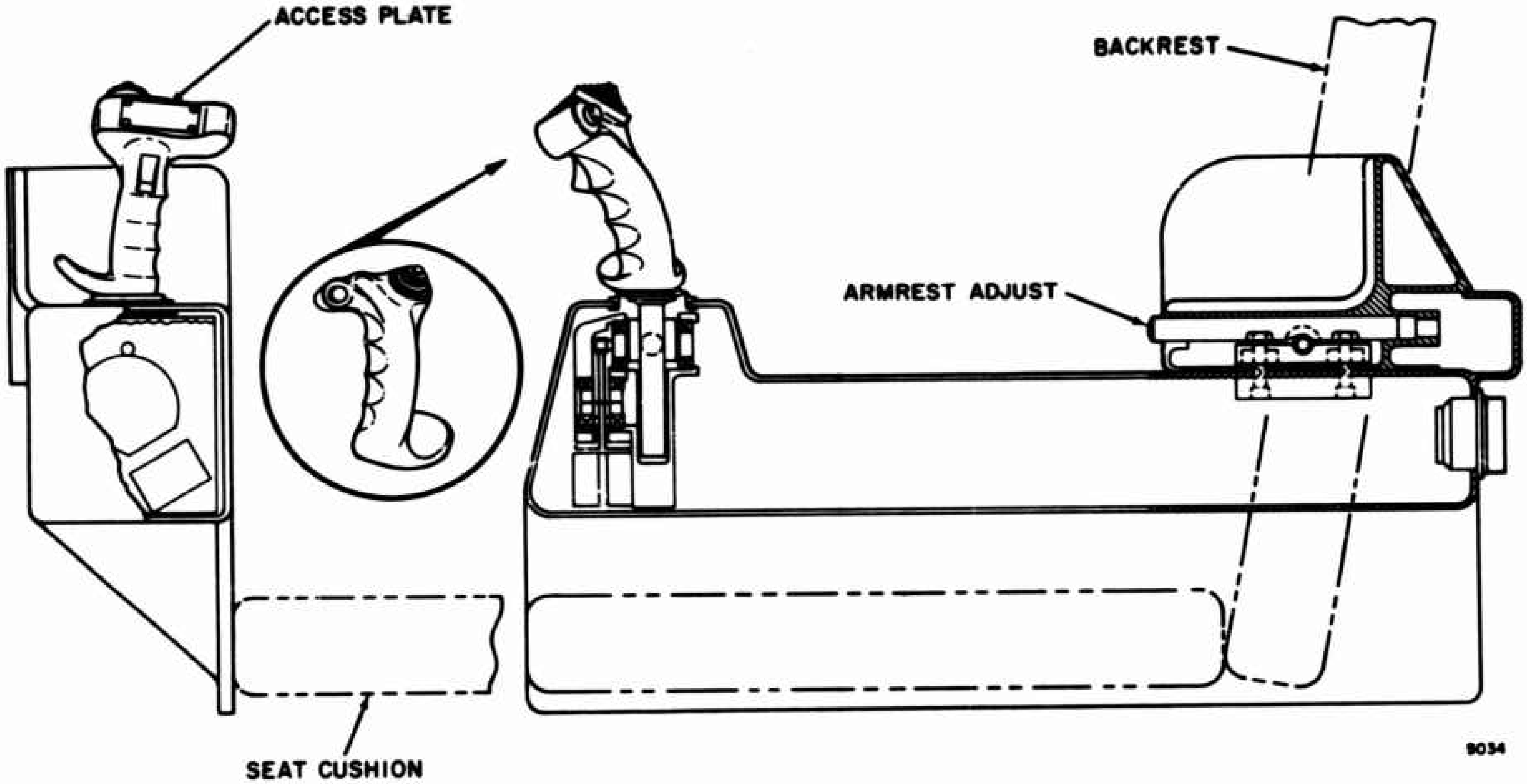


Figure 34
Sidestick Controller

The hand grip is designed specifically for sidestick controller applications. The grip is smaller in diameter than a conventional center stick grip, and provides a deeply contoured thumbrest at the back of the grip for applying precise forward stick motions and conventional finger molds for applying aft motions. The four-position trim switch and the push-button auxiliary switch are located on each side of the thumb indentation. The pushbutton switch is located on the left-hand side of the thumb indentation and the trigger switch is located under the forefinger position. An access plate is located at the top of the stick to provide access to the trim and interlock switches.

The stick grip is mass-balanced to prevent torquing moments about the pivot points during g-loading; damping is provided in all axes of control. The selection of force gradients, pivot location, grip shape, grip neutral position, displacement angles, and switch locations resulted from several military development programs which established these parameters.

The mechanical performance characteristics are shown in table IV.

TABLE IV

MECHANICAL PERFORMANCE CHARACTERISTICS OF THE SIDESTICK CONTROLLER

Characteristic	Performance
Grip	±15 deg pitch ±12 deg roll
Pitch Force Gradient	0.7 lb/deg for 10 deg 1.5 lb/deg from 10 to 15 deg
Roll Force Gradient	0.3 lb/deg for 7 deg 0.9 lb/deg from 7 to 12 deg
Grip Neutral Position	15 deg forward (adjustable) 8 deg inboard (adjustable)
Friction at Pressure Point	0.2 lb (max) at 3 g
Mass Unbalance	8 in.-oz at 3 g (max)
Damping	1/4 sec from first stop to neutral, no overshoot
Pressure Point to Pivots	3 in.
Detent Forces	0.75 lb both axes
Elbow rest adjustment range from pressure point	12.5 to 15.5 in.

3. TRANSDUCERS

Transducers convert mechanical motion or force into proportional electrical signals. They are used for control stick inputs, for actuator feedback and/or monitoring and possibly for flap position or wing sweep position indication. Of primary concern are the control stick and actuator transducers. Two classes of transducers can be employed on the control stick: force and position. A relationship exists between the applied stick force and stick position for commands at a given flight condition. The relationship depends on the force and position cues that pilots prefer for various flight conditions. For example, at high speeds, the force cues predominate and the pilot likes a sensitive stick; at low speeds (approach) position cues predominate and a loose stick is preferred. As long as the proper relationship is maintained, either force or position transducers can be employed. Employing both is an unnecessary complexity because stick compliance can satisfy the required relation. Two types of sidestick controllers have been employed. The first is a conventional gimballed stick which has a centering spring and uses position transducers. The other is a force stick which is firmly attached to its base (i.e., nongimballed) and uses force transducers mounted on a compliant member near the base. The compliance may allow no motion or large deflections. The gimballed stick is more popular because of the motion cues that it supplies at low-Q flight conditions.

Two types of force transducers are in use today. Both have low power output suitable for signal use only. The first type employs an E-core transformer pickoff or an LVDT (linear variable differential transformer) position transducer to measure the deflection of a calibrated spring. The spring is very stiff and is used as a series link with the applied force for direct measurement. It may also measure a fixed fraction of the applied force. The output is linear to within ±2 to 5 percent of full scale over the temperature range. Temperature compensation is difficult because the thermal dependence is nonlinear. The position transducers are inductive devices having ac outputs. Both are designed to measure the very small deflections which are typically ±0.010 inch. Potentiometers are not suitable because they introduce small deadzones due to contact friction.

The second type of force transducer employs a strain gage bridge which is bonded to a strain member to measure the strains induced by an input. The member usually transmits a small fixed fraction of the force, but it may also transmit the full force. Redundancy is very easy to implement since a gage might typically be a few tenths of an inch on a side. Strain gage excitation might be ac or dc. Linearity and gradient stability can be held to within 0.1 to 1.0 percent of full scale. Temperature compensation is relatively simple to obtain because the thermal dependence is linear. Two potential problems of strain gages cause a high failure rate: signal-to-noise ratio and bonding. The best strain gages available today are metal foil. The signal-to-noise ratio of foil and wire gages is very low. To provide a usable output, they are operated near the endurance limit of the gage material, which leads to a high failure rate, or a high gain stage of amplification is used or both. Semiconductor strain gages solve this problem because they have a much larger output. Their resistance is high and the change with stress is large. Foil gage outputs are in millivolts; semiconductor gage outputs are in volts.

The second problem is that of properly bonding the gage to the strain member. Poor bonding causes null creep, long term drift, and poor operation at the temperature extremes. Bonding technology has developed tremendously, but the human element has prevented elimination of the problem. The recent breakthrough in depositing semiconductor gages directly onto silicon strain members will essentially solve the problem.

The choice today between the two types of transducers would depend on the tradeoff between the size and reliability of a redundant transducer. The inductive pickoff and spring occupies a minimum of several cubic inches apiece. This normally precludes installing redundant transducers inside a control stick since at least three are needed in each axis. They would have to be externally mounted. Foil strain gages have at least three times the failure rate of the spring type ($\lambda > 30$ versus 10 failures each 10^6 hours). The initial costs are about equal. Therefore, in a center stick, the inductive pickoff and spring should be used, but in a sidestick strain gages should be used. Semiconductor strain gages will be at least as reliable as the inductive pickoff and spring. Therefore, when these gages become available in the near future, they should be used in either application.

Four types of position transducers are available: the potentiometer, synchro, LVDT, and digital encoder. The familiar potentiometer is a variable resistance device that can be energized by either ac or dc power. It is available in nearly any size, shape, and resistance. Its output can be shaped to a specified nonlinear function with either linear or rotary output. While a potentiometer can be made to control relatively large power levels (such as a few watts), it is generally best suited for signal power level outputs because of the large internal losses. The resolution of wire wound potentiometers is limited by the wire spacing; the resolution of composition potentiometers is unlimited. The newer conductive plastic types are preferred because of their higher reliability although signal noise may be a problem. The reliability of potentiometers is relatively poor; a typical failure rate is at least 100 each 10^6 hours. Nearly all of this rate is due to the action of the wiper contacts which are very susceptible to vibration and wear. Cost ranges from $5 to $25.

The synchro of concern here goes by many names: induction potentiometer, linear transformer, linear synchro, rotary transformer, or rotary variable differential transformer. The device has a single rotor primary and a single stator secondary winding which is normally wound to provide a linear output with rotation rather than sinusoidal. Being a transformer, the windings are electrically isolated and the resolution is practically unlimited. For reliability a brushless synchro should be used. The rotor input comes through the rotor shaft via flexible leads rather than via sliprings, and rotor motion is stopped short of ±90 degrees to avoid ambiguity and lead twisting. Excitation is, of course, 400 hertz at either 26 or 115 volts. Input power ranges from 0.1 to 1.0 watts. Linearity is commonly ±0.5 percent of full scale. Synchros are sized according to outside diameter. The smallest practical and proven size is a size 8 which has a 0.75 inch diameter. Its weight is 36 grams. A smaller size 5 is available with a 0.5 inch diameter, but its reliability is as yet unproven. Output power level is small being measured in milliwatts. Cost ranges from $40 (in large quantities) to $80. Synchro reliability is the best of the position transducers with a failure rate of 8 each 10^6 hours for this

application according to MIL-HDBK-217A. Figure 35 shows a mockup of a quadruplex tandem linear transformer using size 8 devices. It weighs slightly over 5 ounces. The 6-inch device could cause packaging problems. Use of a pair of dual tandem devices in parallel would minimize this problem.

The LVDT is a rectilinear transformer having three windings wound on a hollow core. One winding supplies the excitation flux; the other two are connected in series bucking to provide a differential output. A movable iron core varies the coupling between the excitation and output windings such that at null, the voltages induced in the two secondaries cancel. Displacement from null causes a differential output as the voltage in one winding increases and the other decreases. Resolution is unlimited. Excitation ranges from 5 to 115 volts at 400 hertz depending the size of the LVDT. Linearity is typically ±0.25 percent of full scale. Displacement ranges from ±0.005 to ±5.0 inches. Lengths are between 0.4 and 5.0 inches; diameters run between 0.3 and 0.9 inch. Weight is less than 5 ounces. Cost ranges from $25 to $300. The failure rate of an LVDT is about 12 each 10^6 hours according to commercial airline field data. Power input (and output) of a standard LVDT is less than 1 watt. A special high power LVDT was fabricated for Douglas Aircraft company for their fly-by-wire project (discussed in paragraph III.2.a) which could provide 20 watts of output power for each channel. The triple tandem LVDT was 15 inches long, 3 inches in diameter, and weighed 30 pounds. Total stroke was 3 inches.

Digital encoders are available, but they have already been ruled out of consideration due to their incompatibility with analog flight control equipment. They are rotary devices having from 100 to 620 counts for each turn which establishes the resolution. A 10 to 200 kilohertz interrogation signal is required plus the electronics for readout. Sizes range from a typical size 8 or 11 synchro case to a 2-inch diameter and 3-inch length. Weight is between 0.25 and 1.0 pound. Cost ranges from $200 to $600. The reliability is comparable to a potentiometer.

4. TRANSMISSION LINE

The prime considerations in selecting materials for the electrical signal and power transmission lines are the integrity and weight. Weight, of course, is to be minimized in any flight control system. Integrity is affected by damage caused during battle or maintenance. Battle damage can sever electrical cables (and hydraulic lines) even if they are protected within a heavy conduit. Maintenance personnel can accidentally damage a cable by drilling or cutting nearby in the airframe or by using the cable as a handhold. Damage effects can be minimized by separately routing redundant cables, one for each channel, so that the damage affects only one channel. The physical separation of the cables should be as great as possible. The connectors are also part of the transmission line. Their integrity and reliability are affected mostly by the number of times that they must be disconnected. Each time a connector is removed and replaced, a small chance exists that pins will be bent by forcing a misaligned connector into its receptacle. Cable flexing also causes wear and tear on the wiring. Disconnection is required when a failure occurs in the LRU (line replaceable unit) or when periodic maintenance is needed. The mean time between a required disconnect for an electronics assembly is the inverse of the total failure rate of the LRU. For the fly-by-wire system, the MTBF (mean time between

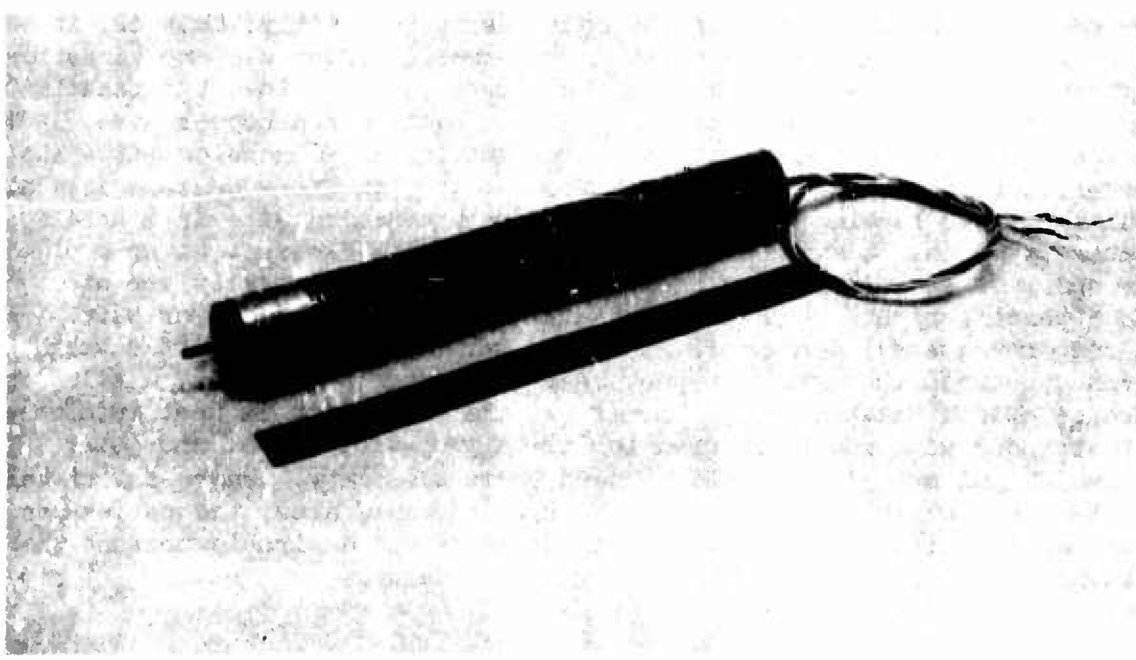


Figure 35
Quadruplex Tandem Linear Transformer

failure) of an electronics channel LRU is estimated to be 14,000 hours; for a control stick transducer LRU it is 25,000 hours, and for an actuator LRU it is 1,200 hours. The connectors on the first two LRU would not likely be exercised more than one to four times during the life of the airplane depending on the class of aircraft. (Equipment in a transport operates for 6,000 hours annually, and the aircraft operates for at least 10 years. The mean lifetime of a fighter might be expected to be 5 years. This gives a total equipment lifetime operating time of 60,000 and 30,000 hours respectively.) The actuator connector would be used 25 to 50 times over the life of the aircraft, assuming no regularly scheduled maintenance. Even this is not excessive. Preflight or postflight check would detect any cable failure before the next flight to ensure transmission line integrity.

Several choices of electrical wire exist for use in aircraft: standard stranded copper, reinforced stranded copper (mixed copper and stainless steel stranded wire), solid copper, and copper-clad steel. Solid copper is the least desirable because of its relative inflexibility; that is, it tends to work harden and become brittle more readily under use and vibration than stranded wire. Copper-clad steel has three to five times the tensile strength and 2-1/2 times the resistivity of an equal sized copper wire. A 16 gage wire is needed to achieve the same resistivity as 20 gage copper. Its relative stiffness would be a deterent for use in connectors because it would be much harder to handle. The reinforced copper stranded wire is a good compromise. It has 16 strands of copper and 3 of stainless steel or 6 of copper and 1 of stainless. Its tensile strength and resistivity are about 30 and 5 percent higher respectively than standard stranded copper wire. Since copper and steel have nearly the same density, there is little weight difference between the various types. Also, in large quantities, little cost difference exists between them. Therefore, the conclusion is that reinforced copper stranded wire should be used for the fly-by-wire signal and power transmission and shielding should be used where necessary. Where additional protection is desirable, such as in a high maintenance area, the cables can be run through conduit. Using conduit everywhere is not desirable because the protection is not needed and it adds a weight penalty.

5. SUMMING JUNCTIONS

Signal summing occurs at two places in the fly-by-wire systems: (1) where the pilot's command is summed with C*, AFCS, and possibly the trim signals; and (2) where the servo command is summed with servo feedback. In the first case, all signals are electrical, and the summing junction is an electrical or electronic element such as an amplifier or transformer. A transformer is unsatisfactory for two reasons. First, in a redundant system where all channels are not excited by the same power supply, signals to be summed may be out of phase or even at different frequencies. This problem of nonsynchronous power supplies has been discussed previously. Therefore, ac summation is not possible. Even if all channels are excited commonly to eliminate phasing problems, the transformer still has difficulty in summing. The transfer impedance is nonlinear since it depends on the flux level in the core. The effect of a command signal then depends on the size of the other inputs, such as the AFCS or trim, at any particular instant and vice versa. The shifting gains would be very disconcerting to the pilot. For example, with no AFCS input, as when it is not engaged, the pilot would get a response considered normal. If instead the AFCS were introducing a command

simultaneously, the pilot would get a lesser response because the forward gain would be reduced. Therefore, transformer summing junctions are undesirable. The best summing junction for use with either synchronous or nonsynchronous power is at an amplifier input since summation can be either ac or dc. Obviously for nonsynchronous power, summation must be dc. The amplifier summing junction appears to be an electrical ground to the input signals because it is a current node. That is, the sum of all currents at the summing junction is zero. Hence, each input which is summed through a gain controlling resistor is electrically isolated from all others; the summation is always linear and independent.

The servo summing junction will be electrical where electrical feedback is used, or it may be a force or position summing device where mechanical feedback is employed. Where electrical feedback is brought out separately to sum with the command signal as is commonly done, the amplifier is the ideal device to employ. In Sperry Phoenix's fail-passive approach to be described later in Section VI, the feedback transducer is the summing junction. The object is to sum the electrical input with the mechanical position feedback in such a way that a transducer failure blocks transmission of the command signal. This eliminates the occurrence of an open loop caused by a loss of feedback. The transducer is a modified resolver which is very similar to the induction potentiometer used for the control stick transducer except that it has an extra stator winding. The two stator windings are wound at 90 degrees so that their fluxes are spatially orthogonal. One winding supplies a constant excitation flux while the other provides the input. The vector sum of these fluxes induces a voltage in the rotor winding. The rotor voltage is proportional to the sine of the angle between the flux vector and the normal to the rotor. The output voltage is the servo error signal. If more than one input comprise the servo command, a summing amplifier is required for isolation as discussed earlier.

Another summing junction that has been used with mechanical feedback is the servovalve torquer. Current into the torquer coils produces a torque that operates the servovalve. Mechanical position fed back through a spring produces a counter torque proportional to position which opposes the command torque. The torques just cancel at the commanded position (allowing for the small offset error required to hold the load at that position). This technique is adequate in nonredundant servos where accurate control is not required and simplicity is desired. The disadvantages are that changes in temperature and the local magnetic field cause offset errors. Further, redundancy is extremely difficult to implement because of the complexity of the multiple feedback links and the almost impossible task of maintaining synchronization and alignment of the channels. The latter problem not only hinders proper operation, but it complicates the monitoring task as well. Therefore, torque summation is not desirable for fly-by-wire systems.

6. ELECTRONICS

The electronics components, circuits, and packaging techniques most suitable for use in fly-by-wire systems are now within the state of the art in automatic flight control system design. As discussed earlier, the

electronics should be primarily dc except for the fail-passive approach which uses ac circuitry. The difference between the two as far as components are concerned is negligible. AC circuitry requires a greater number of demodulators and modulators plus coupling capacitors or transformers. Proven solid-state components and microcircuits should be used to the greatest extent possible to maximize reliability. For example, transistor and optoelectronic switches have replaced signal relays. Solid-state power switches are available for limited power switching only. A good deal of development is being done in this area. Microcircuits both minimize the number of components and the number of connections which improves reliability. Connections contribute a significant portion of the total electronics failure rate primarily because of the effects of the human element during manufacture.

The use of dc electronics in fly-by-wire control provides a number of advantages from the standpoint of simplicity, accuracy, and the ability to be microminiaturized. Such an approach represents a significant departure from the more traditional 400-hertz suppressed-carrier ac control techniques which have previously been used in aircraft flight control systems.

DC control systems have a historically poor reputation in airborne applications. The severe problems associated with maintaining drift free and balanced circuit performance over a wide temperature range plagued the designers of dc control systems in the vacuum tube era. Chopper-stabilized amplifiers could cope with these problems but only with some undesirable penalties in circuit complexity. The advent of transistors led to even more disastrous dc control system design failures, for now the problem of leakage and poorly controlled device parameters were added to the vacuum tube dc amplifier's more simple parameter tracking requirements. It is perhaps a knowledge of these early design failures which has created an attitude of extreme caution and even reluctance on the part of control system designers and users when consideration of dc systems is suggested. Yet, within the past 5 years, progress in semiconductor device and circuit technology in relation to the dc operational amplifier has been so revolutionary that one must reappraise many of the design practices, and perhaps even the prejudices which often dictate the mechanization of control systems.

Ironically, one of the factors which has prevented a greater acceptance of dc control systems is their apparent simplicity. It is easy to understand the operation of such systems, but a good design involves many factors which are too easily overlooked. Some system designers have approached the problem of synthesizing control systems in the same manner as they would program a dc analog computer. They have, therefore, often created modern-day counterparts of the early unsuccessful dc systems. Consequently, even the availability of high quality operational amplifiers has not prevented their misapplication in control system designs. Design factors too often overlooked involve: current drift limitations which restrict resistance ranges, impedance balance requirements which restrict flexibility in changing summing and feedback networks, and certain scaling and grounding restrictions. These factors will be discussed in greater detail in succeeding paragraphs on design considerations. First, however, some of the recent history of Sperry Phoenix dc flight control electronics designs is outlined in the following paragraph.

In 1964, Sperry Phoenix designed and qualified to an extremely severe environment an all-dc, three-axis aerodynamic control autopilot for the USAF Maneuvering Ballistic Reentry Vehicle (MBRV). The very complex, large dynamic range and wide bandwidth control laws, the high precision requirements for these computations, and the stringent size-weight requirements dictated the choice of dc computation and control techniques. In 1965, production autopilots were delivered and as November 1966, over 18,000 operating hours have been accumulated on this autopilot design without a single electronic failure or performance degradation. The primary significance of this performance record is the long term drift stability. This is especially important when it is noted that not a single trim potentiometer was required to balance any control channel. A servo amplifier unbalance, for example, is less than 0.05 percent full scale. Figure 36 shows the reentry vehicle SAS to demonstrate the size of fly-by-wire electronics which would be comparable.

The many benefits of dc computation, however, can be obtained only by careful observation of the limitations of the dc operational amplifier computing and control techniques. Some of these limitations are:

a. Signal limiting cannot be obtained with devices as simple as the variable voltage biased diodes commonly used in ac systems. It is most desirable to implement limiting functions by scaling the problem so that the limit is provided by the inherent saturation limit of the amplifier (about 10 volts - zener diodes can be used to adjust this voltage down to about 6 volts). This requirement imposes requirements on scaling and consequently limits the flexibility in changing limits and control gain. Continuously variable precise limiters can easily be implemented but they are considerably more costly than the variable limits which are implemented with ac systems.

b. The ease of synthesizing any desirable gain by controlling summing and feedback resistors is deceptive. Maximum resistance values are imposed by current drift problems which will be discussed subsequently. One cannot, in general, simply change the value of the summing resistor to change a gain. The amplifier configuration allows summing at either the inverting or noninverting inputs to maintain flexibility over signal polarities. However, minimum drift operation of these amplifiers requires that the impedance to ground from both summing bases be balanced. Consequently, changing or adding a summing network at one base usually requires some adjustment of the impedance at the other summing bases. This represents a penalty in flexibility, but a good system design can make provision for changing and adding inputs without causing any disruption of the basic system building blocks (microelectronic subassemblies).

c. Isolated, regulated internal power supplies are needed for proper system operation. The signal ground must be isolated from power ground of the input power; thus, autotransformers cannot be used.

d. An operational amplifier having a voltage offset of V millivolts cannot in general be balanced by adding -V millivolts at the input through a trim potentiometer connected to a dc voltage reference. This trimming operation may bring the output to zero at a given temperature so that such a procedure is allowable for operating

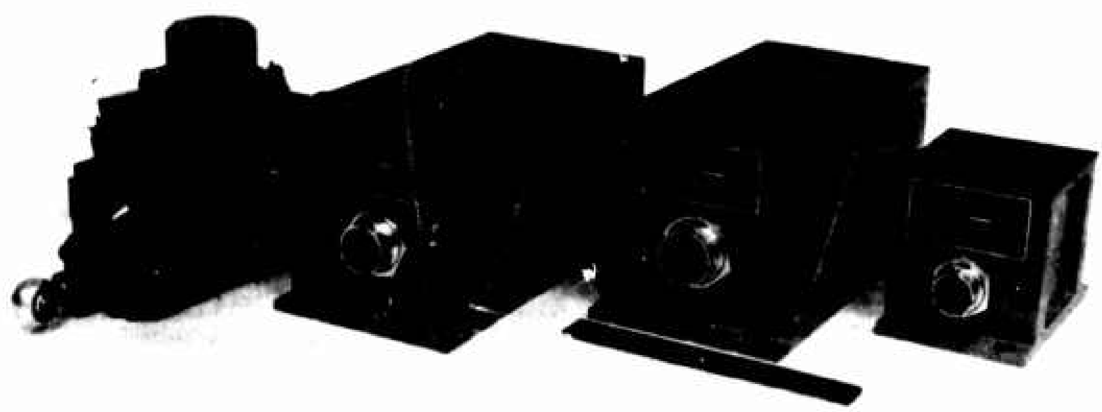


Figure 36
Reentry Vehicle SAS

laboratory equipment. In an airborne control system design the temperature drift effects are usually far more significant than the room temperature offset voltage. Thus, achieving adequate performance over the temperature environment involves design criteria other than simple trim potentiometer nulling. If these criteria are followed, the trim potentiometer is usually unnecessary. The major consideration here involves the allowable range of summing impedances and their relation to the amplifier's inherent current drift.

While it is well recognized that microelectronic integrated circuits have made a revolutionary impact on all aspects of avionics technology, the progress in flight control has been less rapid than in navigation and communication areas. This cautious approach to microcircuitry springs from the unique position of the flight control electronics as a complex and often unbounded collector and producer of signals and data from and to a multitude of diverse aircraft subsystems. Mocrocircuitry finds its optimum application in performing standardized functions. This is the very antithesis of the autopilot's traditional role. Flight control systems interface with ac transducers, dc transducers, ac servos, dc servos, hydraulic control valves, analog instrument displays, discrete command switches, discrete displays, and any other device that may be created to sense, actuate, or display. The electronic functions which a flight control system must perform to adapt the levels of these various signals to the low voltage operating levels of microcircuitry is often a major part of the signal processing functions. Hence, one can never expect an all microelectronic implementation of the traditional flight control functions until all signal and logic interfaces are made more compatible with microcircuitry operating voltage and power constraints.

With the size advantage afforded by the microcircuit operational amplifier, a complete, fairly sophisticated computing function can now be packaged within a single embedded microelectronic subassembly of a standard 0.7 inch x 0.8 inch x 1.2 inches dimensions (figure 37). The computing and control microelectronic subassemblies include both microcircuits and discrete components. These highly efficient modules exploit the size and advantage of microcircuits to the fullest potential consistent with cost and reliability constraints. A typical microcircuit card is shown in figure 38.

Further systems will be expected to employ the hybrid microelectronic packaging approach. This approach is based on hybrid assemblies of microcircuit chips, thick film resistors, and various forms of capacitor devices mounted on ceramic substrates. When the hybrid technology is sufficiently developed where all parts can be mounted to the substrate by flip-chipping, not only will costs and weight be reduced, but the elimination of wire interconnects will constitute a major reliability improvement. Metal oxide semiconductor (MOS) techniques may reduce the size, cost, and weight of hybrid electronic assemblies by several orders of magnitude, once reliability, temperature, and interface problems are solved.

7. ACTUATORS

Actuators may be required in two places in a fly-by-wire system: surface actuators which are hydraulic because they provide higher power and performance in smaller packages than electrical actuators, and trim actuators which

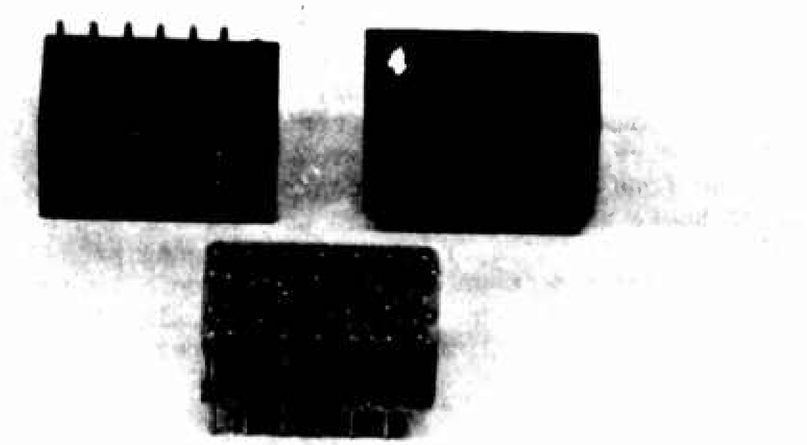
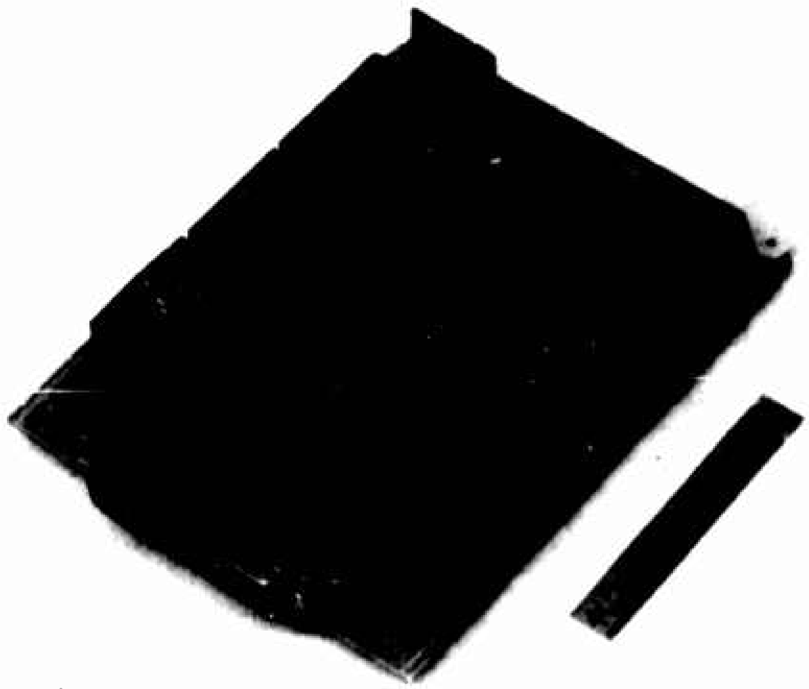


Figure 37
Typical Microelectronic Subassembly



(a) COMPONENT SIDE

(b) WIRING SIDE

Figure 38
Typical Microelectronic Card Assemblies

are electrical because they are more reliable and smaller for the required performance. The hydraulic surface actuators are discussed in detail in paragraphs VI.4 and VI.5. These actuators differ from available actuators primarily in the higher degree of redundancy employed and the attendant design problems. They further differ in that they combine the functions of the series and boost actuators so that a higher-performance power actuator is generally required.

The control stick actuator would be required only if parallel stick motion is deemed necessary for trim or AFCS inputs. Series inputs produce no stick motion. Hence, an actuator is not necessary and the system becomes simpler and more reliable. The power and performance requirements of the parallel actuator are low enough that small electromechanical servos are adequate. The trim actuator would position the neutral point of the stick centering spring in response to commands from either the pilot's trim controls or to rate commands proportional to the long term average actuator error signals. The unit would consist of at least two servo modules, each containing an integrating servo, and an associated electronics module. The servo integrator includes a tachometer, gear train, and clutch. A typical actuator would employ a standard size 11 or 15 servo motor. Now a size 11 motor can provide 30 to 40 pounds stall force and a free speed of 0.5 inch each second at its output which is clearly adequate for this application. The weight of a typical dual unit would run about 8 or 9 pounds and have a volume of 250 to 300 cubic inches.

8. AIRCRAFT MOTION SENSORS

Artificial feel can be implemented in a number of ways as discussed in Section III including using either C* feedback or dynamic pressure q to control system gains. Deriving C* requires attitude rate and normal acceleration sensors; deriving q requires a sensor that takes the ratio of pilot-static to total pressure. A summary of sensor requirements and available types are listed in tables I and III.

The only proven rate sensor available today is the single-degree-of-freedom, spring-restrained rate gyro. Many types of rate sensing techniques are being developed having the primary purpose of eliminating gyro spin bearings and the wearout problem (95 percent of all gyro failures are due to rotor bearings). Several concepts show promise, such as the ring laser and the use of air bearings, but none are expected to be available and proven within the next 2 years. The factors to consider in a rate gyro are whether or not it is reparable, the type of pickoff, the monitoring provisions, and expected life. A reparable gyro initially costs nearly twice as much as a nonreparable type, but its overall costs are lower because it can be overhauled. While the nonreparable types are smaller, this is actually no advantage because the smaller size reduces heat transfer. Hence, the failure rate of a nonreparable is often higher than a reparable one. A nonreparable gyro is typically 1 inch diameter x 2-1/2 inches long and weighs 8 ounces; a reparable type is typically 1-1/2 inches diameter x 3-1/2 inches long and weighs 14 ounces. The reparable type is slightly larger to facilitate overhaul, but this allows incorporation of other features to improve reliability. In particular heat transfer is improved which lengthens bearing life. The largest factor in bearing life is failure of the lubricant.

Larger bearings help maintain lower temperatures, reduce lubricant film stresses, and reduce shock susceptibility. This problem is also minimized by using lower speed rotors (12,000 rpm versus 24,000 rpm conventional) and by incorporating reservoirs to ensure an adequate supply of lubricant. Bearing life can be increased by an order of magnitude in this way. The gyro should also be helium filled to minimize lubricant contamination.

The gimbal pickoff should be an inductive device to eliminate wiper contact as discussed in position transducers. A microsyn is very popular because it has no brushes or wipers, and it has excellent sensitivity. Potentiometers are used in some rate gyros; these types should be avoided. Self-test and monitoring of gyros is done in two ways other than by comparison with another gyro. The simplest method is to add a small pulse generator to the rotor so that it generates a pulse each revolution (or half revolution). Wheel speed can be measured in this way. A gyro rotor has the peculiar characteristic that it will almost never quit running once started, but if it has failed it will not start. Therefore, a wheel speed detector will detect 95 percent of gyro failures. For a thorough gyro test such as during preflight checks, an input torquer should be available to exercise the gyro much as a rate input would do. Performance can be judged by comparing the output for a known input with a predetermined reference response. This tests gimbal freedom, damping, torsion restraint, and pickoff. In most self-test schemes, the torque is applied to the gimbal. This checks everything but the rotor. Another type of self-test device applies a force to one end of the rotor to develop a torque on the gimbal. Flux from a small winding acting on the spinning rotor produces an eddy current drag force. Since a test input current produces a torque just as input rate would, all functional elements of the gyro are tested including the rotor speed. The rate gyro characteristics required in a fly-by-wire system are very similar to stability augmentation requirements. They are listed in table III.

The accelerometer can be chosen from two classes: (1) spring-restrained, viscous-damped devices that measure acceleration directly, or (2) internally servoed devices in which the acceleration is indirectly measured by the torquer signal. The latter type are much more precise and are used extensively in inertial systems. They are more complex and costly than fly-by-wire applications can justify. The former are also proven devices but are less expensive. Self-test can be supplied on either type. As in the gyro case, the torquer or output transducer should not have brushes or wiper contacts for high reliability. Those types of accelerometers employing potentiometer pickoffs should not be used. Accelerometer characteristics required for fly-by-wire applications are listed in table III. The lateral accelerometer is included for possible future use in lateral C* command systems.

A simple q-spring consists of two chambers separated by a flexible (zero spring rate) diaphragm. One chamber accepts pilot pressure while the other accepts static pressure. The flexible diaphragm can function as the spring output or it can serve as the input to a servo which provides the actual restraining stick force. In the latter case, the q-spring becomes a small air data sensor rather than a large spring. This approach has more merit for use on electrical sticks because of its smaller size even though

it has more parts. In an installation that already requires a parallel servo, the actuator may be able to serve both purposes. The sensor would be very similar to the servoed force-balanced aneroid cells presently being used in most air data computers. Such computers are known for their unreliability. Several companies, including Sperry Phoenix, are developing solid-state air data sensors to overcome the reliability problem. Solid-state sensors should become available by 1968.

SECTION VI

SYSTEM DESIGN

1. SYSTEM DESIGN CRITERIA

Thus far in this report we have described the problems that exist in modern mechanical flight control systems that prevent achieving their performance objectives. We have described the fly-by-wire approach to its solution. We have also discussed the component selection available for its construction. Before proceeding further, we must establish the flight control system constraints and design criteria around which we must work. The design criteria may be broadly categorized as to performance, reliability, cost, and maintainability. The performance criteria includes not only handling qualities and path control but weight and volume as well. The design requirement is generally speaking to be able to meet the same handling qualities and path control as the mechanical systems according to military specifications. The system must also match or exceed the weight, volume, reliability, cost and maintainability of present-day mechanical designs. We have demonstrated by example and comparison that the performance, cost, and maintainability criteria can be satisfied.

The reliability criterion employed is the accident rate for commercial airliners attributed to the flight control system as established by Kaman in their study. The data were derived from the maintenance records of the Civil Aeronautics Board and the Federal Aviation Agency for the period of 1952 to 1959. The probability of a flight control system failure for a 1-hour flight is 2.3×10^{-7}. This value establishes our reliability criterion. It is more stringent than many military aircraft can meet as evidenced by data from older aircraft. For example, the failure rate employed by Douglas Aircraft in their study was based on the AD Skyraider pitch axis failure rate of 6.14×10^{-4} for a 1-1/2 hour flight.

2. SYSTEM CONSTRAINTS

The system constraints which establish the outer boundaries of the design problem result from the flight control system's environment (available space and power, temperature, vibration, shock, humidity, and so on) and the state of the art not only in components and materials but knowledge as well. Environmental constraints relate to the aircraft while state-of-the-art constraints relate to the control system.

Space constraints are most important to the actuator configuration. The higher degrees of redundancy called for in fly-by-wire systems generally produce larger packages; hence space limitations will be important. Available power is no more a constraint for fly-by-wire than mechanical since the actuation requirements are essentially the same, and the solid-state control electronics cause relatively small power drain. However, the number of supplies is something of a constraint because the implementation of an odd degree of redundancy (e.g., triplex) with an even number of power supplies (e.g., dual) and vice versa is inefficient particularly at the actuator. Operating triplex actuators from dual supplies requires special switching.

Temperature constraints limit the selection of materials or equipment location. This constraint normally affects only supersonic aircraft traveling over Mach 2 where aerodynamic heating begins to cause problems. For example, at Mach 3 the aircraft skin temperature is around 450° to 500°F (232° to 260°C) which is well above the tolerable environment for electronics as well as some hydraulic fluids and plastics. Temperatures above 160°F (71.1°C) are detrimental to rate gyro life.

Vibration and shock are detrimental to everything. Location, orientation, and packaging are constrained by these parameters. Long tandem actuators, for example, are greatly affected by vibration. Vibration can shorten gyro bearing life substantially.

An additional constraint appears primarily in high performance aircraft where control surfaces are very effective and aircraft response is fast. The constraint is the maximum allowable aerodynamic load imposed on airframe by failure-induced transients or excessive commands. The use of C* commands eliminates the latter. Failure transients result from relatively slow detection and channel switchover times during hardover failures. Estimates of the least upper bound on detection plus switching time range from 30 to 50 milliseconds. For very large aircraft such as the C-5A, this value may be as high as 0.5 second.

The control system constraints due to limitations in the state of the art of knowledge and components and materials are slowly receding. Knowledge lacks in the areas of handling qualities, efficient redundancy implementation, and the ability of engineers to work in multiple disciplines. High reliability components are needed particularly in the apparently neglected areas of electrical and hydraulic power generation and in electrohydraulic servovalve and engage solenoid design. Compatibility of components has already been discussed.

3. ADDITIONAL CONSIDERATIONS

Aside from the system requirements, criteria, and constraints just discussed, some additional considerations must be kept in mind during the design process. One is the desirable interchangeability of components between aircraft which is a fringe benefit of the fly-by-wire approach. This greatly improves maintenance and logistics. However, VTOL aircraft will very likely require a different controller than a conventional aircraft because VTOL aircraft have different requirements at hover, and the blending from hover to cruise complicates the design. While interchangeability amongst conventional aircraft will be high and likewise for VTOL aircraft, interchangeability between conventional and VTOL aircraft will likely be low.

Fast maintenance requires failure reporting and BITE as well as good access and simple replacement. This also enhances short preflight checkout time which is becoming a must for combat crews. A 10-second checkout time should be a practical goal. Control system maintenance will probably occur more often in a fly-by-wire system because of the high degree of redundancy necessary to achieve the desired system reliability. Good accessibility, maintainability, and failure reporting, however, will produce aircraft down time equivalent to or better than those obtained with a mechanical control system.

To minimize battle or maintenance damage, the electronics should be packaged by channel (i.e., all channel A's packaged together, all channel B's together, etc) and located separately. The redundant cables should also be routed separately. To provide the desired system reliability, the fly-by-wire control system must be capable of continued operation after the occurrence of two failures. This will be discussed in the next subsection. When a third failure is indicated, the system should fail to a center position or locked to a predetermined trim position. These statements cannot apply to an aircraft having only two electrical or two hydraulic power sources since failure of both hydraulic or both electrical supplies would produce control system failure. The fly-by-wire control system must operate with undegraded performance after a single failure. A second failure may produce limited degradation in performance, but it should be such that safe flight of the aircraft is not impaired.

The monitoring circuits should be capable of reporting failures to the pilot's station not only to provide maintenance information but also to inform the pilot of control system status. The monitors must be either fail-safe, that is, report their own failures, or demonstrate reliability of such a degree as to not degrade system reliability when placed in series with the system. These monitors and switching circuits provide automatic selection of operational control channels for the first two failures.

A fly-by-wire control system should have in-flight reset capability from the pilot's station to be able to clear false alarms. In addition, the monitoring and switching logic should be designed to allow the pilot to select any control channel after a third failure has occurred. This additional logic could result in a three-fail-operate system with pilot select upon the third failure if the system is quadruplex or if the remaining channel is not the model in the triplex system.

4. TRADEOFFS

a. System Tradeoffs

Many of the tradeoffs to be considered in a fly-by-wire design have already been discussed. Artificial feel techniques were described in Sections III and IV from which the C* command approach was chosen. The tradeoff between parallel and series trim cannot be made because it depends primarily on the human factors problem of whether or not parallel stick motion is required. On the basis of reliability and simplicity, series trim is preferred because it eliminates an actuator. Section V contains the tradeoff discussions concerning the control stick, transducers, signal types, transmission line, summing junctions, aircraft motion sensors, and the electronics. The remaining tradeoffs to be made concern the degree of system redundancy to employ and the actuator configuration tradeoffs.

b. Degree of Redundancy

Without doubt, unless the fly-by-wire system design employs redundancy, its reliability will never meet or exceed that of current mechanical systems. The reliability criterion, or rather the probability of failure criterion, has been established as 2.3×10^{-7} each hour. We can relate this

value to the degree of redundancy required by considering the reliability equations for an example control axis having different redundancy implementations. This is best done by considering the unreliability or the probability of failure Q(t) rather than the reliability R(t) because the calculations are simpler and more meaningful. These parameters are related by

$$R(t) + Q(t) = 1$$

The parameters are a function of time because the equations are nonlinear for redundant systems. The following table lists the various redundancy implementations and their probabilities of failure assuming random failures and perfect monitors. The assumed failure rate for each hour λ for our example channel, is 10^{-3}. (The actual failure rate might range between 0.4×10^{-3} and 10^{-3}.)

Number of Channels That Must Operate For System Success	Q(t)	Q(1 hour)	MTBF	Channel λ to Just Meet Criterion
Single Channel	λt	10^{-3}	1000	7.7×10^{-8}
One of Two	$(\lambda t)^2$	10^{-6}	500	2.8×10^{-4}
One of Three	$(\lambda t)^3$	10^{-9}	333	4.3×10^{-3}
Two of Three	$3(\lambda t)^2$	3×10^{-6}	333	1.6×10^{-4}
Two of Four	$4(\lambda t)^3$	4×10^{-9}	250	2.7×10^{-3}

The value for Q(1 hour) must be compared with a third of 2.3×10^{-7} or 7.7×10^{-8} since we are talking about a single axis. The fact that becomes immediately apparent is that the systems meeting the criterion must be capable of tolerating double failures; that is, the system must be able to operate on one of three channels or two of four channels. The MTBF is the inverse of the total failure rate. It provides a measure of the mean time to the first failure when starting with a completely healthy system. It is therefore a measure of the relative complexity and of how often maintenance actions are required. Ideally, the triplex system would be the choice because of its lower complexity. However, our simplified example does not take into account the added monitoring equipment needed to be able to correctly identify and switch out the channels. Depending on the particular system, the extra equipment more often than not amounts to a fourth channel. Other factors not considered are the degree to which the monitors are imperfect (i.e., cannot detect all failures or fails without indication so that a subsequent channel failure goes undetected) and the presence of common elements (such as an output member) with small but finite failure rates which can cause system failure. The last column in the table represents that failure rate which an ideal channel must have such that the system will just meet the reliability criterion. Again, the system that can tolerate double failures can easily meet this requirement since a practical channel including the C* sensors would have a λ ranging from 4×10^{-4} to 10^{-3} depending on the type of components used. Note that with a factor of four reliability improvement, the dual system would also meet the criterion. A factor of six

improvement would be necessary for the two of three system. Therefore, assuming that the reliability criterion did not change and channel reliability were to improve by four to six times in the near future, then a fail-operational system would provide adequate reliability. It would also be less complex, smaller, lighter, and require less maintenance.

Another approach to redundancy is the optimizing technique called fail-passive design which was described in Section III under Related Work. With this technique the triplex system is optimum because little or no added monitoring equipment is necessary. In fact, a dual fail-passive system will come very close to meeting the reliability criterion today.

c. Actuator Tradeoffs

The design of actuators which will operate after double failures involves a large number of parameters and tradeoff factors. The development of such actuators is important to the practical application of fly-by-wire control. The important design parameters and factors (all of which are to be minimized) are as follows:

(1) Failure rate (including false alarm rate)
(2) Switching time and transient caused by a failure
(3) Cost (including initial and maintenance)
(4) Size (volume)
(5) Weight

These parameters are all relative except the second for which an absolute (even though somewhat subjective) standard is defined. The standard system for comparison is a conventional nonredundant actuator. The maximum switching time criterion is based on a maximum allowed normal acceleration or displacement, which is regarded as unsafe, at the flight condition for maximum control effectiveness (e.g., Mach 0.9 at sea level) for the aircraft in question. The time is measured from the onset of a hardover failure to the time when normal operation (i.e., proper actuator output) is restored. This includes the time to detect the failure and the time to switch out the failed channel. Switching in a new channel, if required, is assumed to occur simultaneously with switching out the failed channel. The selection time may range up to 300 milliseconds depending on the class of aircraft. A time of 50 milliseconds is assumed for this study.

The search for suitable actuator designs is hampered by the great number of tradeoff factors involved which include the following:

(1) Degree of redundancy (dual, triple, quadruple, etc)
(2) Type of redundancy (active or standby)
(3) Position or force summation
(4) Type of servovalve (flow or pressure control)
(5) Hydraulic or electronic monitoring
(6) Secondary actuator (yes or no)
(7) Mechanical or electrical feedback

Additional factors complicating the design are the:

(1) Number of power supplies (two, three, four, ...)
(2) Available space for the actuator

The number of possible combinations is in the thousands. However, by investigating these factors more closely, we can eliminate some of them or at least minimize their influence on the design thereby reducing the number of combinations to a tractable level.

First of all, the degree of redundancy employed is related to the double fail-operational requirement. Triplex actuators are awkward to design for two or four hydraulic supplies, and duplex or quadruplex actuators are awkward to design for three supplies. To meet the reliability requirement, at least three channels are required to ensure that at least one channel remains operational after two failures. Further, to ensure that the actuator fails to neutral (or trim), a fourth channel is needed for comparison with the third. The fourth channel may be real or simulated (this is a minor tradeoff factor not listed above). This means that in one form or another, a minimum of quadruplex redundancy is required in the actuators. Note that since the electrical and hydraulic power supplies are independently monitored, three supplies are needed; systems with dual supplies will compromise the flight control system reliability unless the supply reliability is improved significantly. The failure rate of a hydraulic power supply is typically one each thousand hours. Therefore, the probability of two hydraulic supplies failing within a 1-hour flight is 10^{-6}. This failure rate is higher than the control system itself.

In active redundancy, all channels are in the control path; in standby redundancy, one channel is active while all other channels are operating but not in the control path. Active redundancy is preferred over standby, particularly in the actuator, for two major reasons. First, a transient caused by a failure or in switching out a failed channel is minimized by the opposing actions of the other channels. Second, the packaging efficiency is better in an active system because the actuators are sized such that at least two can carry the load. In standby redundancy, however, each actuator must be sized to carry the load thereby requiring actuators with at least twice the capability as before. The standby redundant actuator will then be larger and heavier by approximately 30 percent. The primary reason for using standby redundancy is to maintain constant performance after one or more failures. This is another important consideration where a surface position is commanded as in most control systems. If in a high-Q condition an excessively large position is commanded, stresses on the airframe or the surface may be excessively large. Therefore, the actuator is designed with a force limit so that the large position cannot be commanded. Now in a C* command system, the pilot commands an aircraft acceleration or rate rather than surface position. The surface goes to whatever position is required for the maneuver. The C* command system protects the airframe by limiting the maximum commandable acceleration. Since the surface actuator is now merely one component in the forward path of a high gain control system, its performance need not be held constant. Degradation or variation of performance due to failures has relatively little effect on the performance of the overall system (in the linear range). Because of the high forward gain and control effectiveness at high-Q flights, large surface displacements do not occur even for abrupt maneuvers. Therefore, aircraft loads

will not be excessive. It should be pointed out that local stress levels are not controlled directly by C*. In a triplex active system, the actuator is sized so that any two channels can satisfy the force requirements. Then, under normal conditions, a 50 percent excess will be available. This level should not overstress a control surface. However, if the maximum force output of any active redundancy actuator exceeds the designed safe level for a surface, then the safe level will establish the maximum allowed output. The resulting actuator can still provide the required performance with one failure in a C* command system and satisfactory performance under double failure conditions.

Two methods of summing the outputs of redundant actuators are by force and by position. Force summation combines actuator outputs through a rigid link so that relative motion between the channels does not occur. Examples would be connecting actuator pistons in tandem on a common shaft or in parallel into a torque tube. The actuators then are all forced to have the same position and velocity, and the load force is shared. Position summation allows relative motion between channels through a "walking beam" link. Under normal conditions both techniques operate identically, but under various failure conditions, the differences become apparent. The four types of failure conditions are hardover output, free or passive, hydraulic lock, and seizure or jam.

In force summation, a failure reduces the maximum force available, but the position and velocity remain unchanged. This assumes that a hardover failure or hydraulic lock is bypassed and eliminated. A seizure or jam fails the entire system since all outputs are locked. Fortunately, actuators can be designed with no metal-to-metal contact so that the probability of seizure is essentially zero. A hardover force in one channel is opposed by the other channels as soon as sufficient output motion occurs to generate opposing error signals. Because of the high servo gains normally used, the output error is very small (e.g., 0.1 percent of full stroke). This action also naturally minimizes transients.

In position summation, a failure generally reduces the output stroke and velocity (e.g., in half for a dual system), but the force output remains unchanged because the output moment arm is half the actuator moment arm. The position gain must be increased to regain the proper output. This leaves a system with normal position gradient and force but with half its former rate or frequency response. A seizure or hydraulic lock does not affect performance (unless it occurs off center position) since the failed channel must be locked in any case. A free failure fails the system unless the failed channel is centered and locked because no output motion can occur. The good channels oppose a hardover position failure in one channel by going hardover in the opposite direction. The net result is a small position offset from the neutral position. However, the size of the transient depends on where the position transducers are located. If they are located after the summing linkage, the opposing effect is immediate since the system automatically tries to maintain the commanded position. If they are located before the summing linkage, the correcting or offsetting error signal does not develop until after the aircraft responds to the failure and a C* signal develops. Therefore, in the latter case, a significant transient could develop. Failure detection depends on information from individual channel positions and velocities (servo error signals). Position summing has been

popular because the availability of both servo positions simplifies monitoring. However, for higher degrees of redundancy, all positions may not be separately available without using an extremely complex summing mechanism. For example, a practical quadruplex system could likely sum two dual tandem actuators. Monitoring actually becomes more complex than in force summing because position data alone do not allow unambiguous detection of the failed channel unless each channel in independently monitored. Position summation has also been used in the past in dual systems for packaging considerations because the actuators are side by side wheras force summed actuators were commonly end to end. However, higher degrees of redundancy make position summing mechanization very complex, and force summing into a torque tube now provides a simple side by side arrangement. The conclusion drawn from this discussion is that the fly-by-wire active redundant actuators should be force summed.

No clear-cut choice exists between the use of pressure control (single-stage) and flow control (two-stage) servovalves. A pressure control valve generally serves as the first stage of a flow control valve. The second-stage spool valve (sometimes two spool valves operate in cascade) is a power amplifier to provide high performance output. The power stage can be operated separately from the valve as a secondary actuator which is discussed later.

The servo monitor consists of comparators and logic circuits which detect failures and determine which channel has failed so that it can be disconnected. The function can be mechanized with either electronic or hydraulic techniques. Electronic monitoring is the more proven technique. It utilizes signals from electrical position transducers on the actuator output and/or valve spools, differential pressure transducers across the actuator piston, or the servo amplifier error signals. Electronically simulated channels are also employed. Signals within a channel can be correlated, and signals between channels compared by electronic circuitry. The electronic logic disengages a channel through electrohydraulic solenoids which operate hydraulic engage valves and locks. Hydraulic monitoring is relatively recent (being presently used in the F-111 series actuators). Hydraulic valve spool and actuator position transducers are now being used with spool valves acting as comparators and logic. Conceivably, future techniques would include using differential pressure sensors and fluidic comparators and logic; this would eliminate the use of moving parts and improve reliability. A model channel, when used, may consist of a real channel that never drives the load. It supplies a voting reference for the working channels. The logic drives the engage valves directly.

A relative comparison of the two techniques shows the following. Electronic monitoring has advantages of flexibility, having fail-safe comparators and logic, being easily made redundant, and smaller size and weight. Its primary disadvantage is that the components in the electro-hydraulic interface are relatively unreliable compared to the electronic or hydraulic components, and they add undesirable time lags to channel switching. For instance, the switching time of presently available solenoids is at least 25 milliseconds.

The primary advantage of hydraulic monitoring is its fast channel switching time, which can be less than 10 milliseconds, because the electro-hydraulic interface does not exist. This also eliminates the need for electrical power for switching. On the other hand, the fast dynamic response of the comparators plus the relatively small thresholds used force a requirement for tight electronic tolerance controls to ensure that the channels track thereby minimizing the false alarm (nuisance trip) rate. Channel matching could ease the problem but this approach is not desirable because of the difficulty in maintaining a dynamic match. The use of higher degrees of redundancy further magnifies the problem. The result of the tight tolerance requirements is to increase costs, both initial and maintenance.

The use of a secondary actuator to drive the boost or power actuator is a tradeoff against driving the actuator directly with a two-stage (or three-stage) servovalve. The secondary actuator evolved from the familiar series servo and boost servo combination. The stall force of the secondary actuator will range between 200 and 600 pounds in most applications. This force is equal to or greater than the pilot input force on present boost actuators in mechanical control systems. The mechanical feedback linkage is redundant, and it can be contained internally and sealed in hydraulic fluid. This technique is currently being used on some nonredundant actuators. One prime reason for using the secondary actuator is to move the monitor point away from the servovalve spool. Attaching a position transducer to the spool reduces the frequency response of the valve and complicates the design. For example, to attach an LVDT onto a spool either requires a cavity at one end of the spool to house the armature, or it requires a seal to accommodate an external mounting. The seal adds an undesirable friction force. The cavity causes an unbalanced flow gradient; adding a cavity on the other end of spool balances the gradient. However, the presence of the two cavities plus the LVDT slug inertia reduces the high frequency response of the valve.

Adding a secondary actuator eliminates these problems while providing a mode for monitoring actuator rate. Several additional advantages accrue from employing a secondary actuator. First, the channels of the tandem spool valve driven by the secondary actuator can be readily synchronized to eliminate fighting between the redundant actuators. This allows use of active redundancy. Second, a center end lock mechanism acting on the relatively low power secondary actuator rather than the power actuator can be very small and light yet remain effective. Third, while a two-stage valve could very well drive the secondary actuator, a simple single stage will be adequate for most applications. In this case, the secondary actuator and tandem control valve combination can be thought of as being a variety of a second-stage spool valve. This allows simplification of the servovalve design. While it is true that driving the actuator directly by the valve eliminates the mechanical linkage and secondary actuator, synchronizing the channels is not an easy task. The synchronization mechanism adds undue complexity and reduces the independence of channel failures because the channels must be interconnected. Therefore, the actuator must employ standby redundancy to eliminate fighting. This means that each actuator must be sized to carry the load, as discussed earlier.

The tradeoff between electrical and mechanical feedback is real when nonredundant or even duplex actuators are considered. However, when triplex or quadruplex (or greater) actuators are considered, the complexity of mechanical feedback and its synchronization problems when compared to the relatively simple and flexible electrical feedback virtually eliminate mechanical feedback from further consideration.

Before summarizing the tradeoff factors, several additional considerations primarily concerning monitors must be weighed. Several monitorless control system schemes have developed in which the system tolerates one or more failures without the benefit of monitors thus simplifying the design. Notable amongst these are the NAA Autonetics "Tri-Safe" and the Sperry Phoenix "fail-passive" design. Although these designs can operate without monitors, the failure reporting requirements demands that monitoring be employed.

Two general classes of monitoring are available: in-line and comparison. In-line monitoring employs a test signal, usually continuous, to which the control system will not respond yet which can be traced through the system or group of components to test signal path continuity. The advantage of this technique is that an additional model channel is not needed for comparison. This scheme is applicable to sensors (e.g., rate gyros and accelerometers) and electronic circuitry, but it is not within the state of the art for application to servo monitoring. Although in-line monitoring primarily checks continuity, it cannot detect large drift or gain changes. The signal frequency would be above the control signal frequency band and would likely be sampled before the actuator output (e.g., at the servovalve spool) because the actuator would filter it out. Now, if and when the valve should saturate because of a momentarily large error signal, which would be frequent during fast maneuver such as terrain following or in bumpy air, the valve spool would go against its stops and the test signal would disappear. The monitor would interpret the loss of signal as a failure. Hence, the false alarm rate would be ridiculously high. The in-line monitoring concept was investigated during the project to verify the conclusion. A small series actuator having an LVDT on the valve spool was operated in the laboratory under simulated normal conditions. A continuous high frequency test signal, ranging from just above the actuator's 20-hertz response limit to 100 hertz, was injected along with the command signal during one test and into the actuator position transducer during another. Observing the test signal at the spool LVDT verified that the signal disappeared often and that a false alarm would occur each time. Therefore, in-line monitoring will not be considered further for application to actuators. Although comparison monitoring suffers the disadvantage of requiring an extra control channel for references, it is capable of detecting all failures causing differences between channels above a preset threshold. Therefore, it should be used in all actuator configurations.

The monitor, which consists of comparators and channel select and switching logic, should be so designed that a failure within itself cannot prevent switching out a failed channel. An undetected monitor failure can be particularly dangerous in a standby redundant system wherein the active channel can fail hardover without being opposed. Hence, the comparators should be fail-safe (i.e., self-indicating of failures) so as to properly arm the logic. A good rule-of-thumb criterion for monitor reliability is that it be at least 10 times as reliable as the servo that it is monitoring. Extending this criterion to the system level, the probability that the monitor will cause a system failure (through an inability to detect a failure as well as causing a system failure) should be one-tenth the probability of a system failure. This means that the comparators, logic, solenoids (if required), and engage valves should be fail-safe or meet the system reliability requirement.

The monitor points in any control system are selected so as to minimize failure transients and component tolerance accumulation. For convenience, a point just prior to a summing junction is often used since it is the last point where the unadulterated signal can be observed. Failures between monitoring points are isolated from failures between any other monitoring points. Therefore, points are used to isolate failures. For example, the fly-by-wire system using C* feedback should be monitored at the actuator and at the shaped outputs of the C* sensors just before they are summed with the command signal or before they enter the servo inputs. This isolates the sensor failures from the actuator failures thereby improving reliability and easing the monitoring tasks. Monitoring points associated with BITE are selected to isolate failures to a line replaceable unit, but this is a maintenance aid and does not affect reliability. Actuator monitoring can be performed at a number of places: at the error signal in the servo amplifier, at the servovalve spool position, at the actuator (either secondary or power) position, on differential pressure transducers across the valve or actuator, or on output force transducers. The parameters observed are actuator rate, position, and/or output force. The best monitoring point (or points) will depend on the actuator configuration, but the location should meet the following requirements. To minimize transient outputs, the output member should not have to move for failure detection. This normally means that the primary monitoring point should not be at the actuator output. A corollary to the rule is that the monitor point should include as much of the forward path (excluding the output member) as is possible. The result here is rate monitoring because the last stage before the actuator is nearly always a flow control stage. Spool position monitoring is popular for this reason. Load variations should not affect monitor performance. This rule affects standby redundant configurations or those using model channels. The monitor can interpret a position or rate variation due to a large load on the active channel as a failure because the active and standby channels no longer track each other. Monitoring at more than one point may be desirable in some cases for added confidence by correlating signals. This approach would be helpful in a noisy signal environment.

5. ACTUATOR CONFIGURATIONS

A number of known actuator configurations have been investigated which are capable of operating after double failures according to the reliability analysis. The seven configurations listed do not necessarily represent an exhaustive survey, but data on such actuators are extremely scarce. A large number of configurations exist that are fail-operational. However, the designs generally are not amenable to extension to high degrees of redundancy so they are not considered here. Because the need for an operational capability after two failures has just recently been established, very little development work has been done.

The configurations investigated represent a reasonable cross section of thinking on actuator design. This allows a general comparison of advantages and disadvantages. A direct comparison of size, weight, cost, and reliability was not done, because the concepts are somewhat idealized. The seven configurations are as follows:

(1) Model 1. Conventional standby-redundant actuator with electrical monitoring on valve spool position

(2) Model 2. Conventional standby-redundant actuator with hydraulic monitoring on valve spool position (Hydraulic Research)

(3) Model 3. Secondary actuator with standby redundancy (Weston)

(4) Model 4. Fail-passive secondary actuator (Sperry Phoenix)

(5) Model 5. Standby-redundant actuator with electrical position monitoring (General Electric)

(6) Model 6. Standby-redundant actuator with hydraulic position monitoring (General Electric)

(7) Model 7. Force-summed voted actuator (Elliott Brothers)

The configurations are evaluated on the basis of the tradeoff factors discussed in the preceding subsection.

a. Model 1. Spool-Monitored (Electrical) Standby Redundant Actuator

The model 1 configuration (figure 39) is described first since it utilizes the most familiar concepts. The configuration uses three real channels (identical within tolerances) and a model channel which may be hydraulic or electronic. One real channel is active while the other two operate in standby. Actuator position feedback is electrical. The two-stage servovalves are coupled to the actuator through a four-position engage valve (or through 2 two-position engage valves). The engage valve transfers the system through its operational modes on commands from the electronic monitor via electrohydraulic solenoids. Position transducers on the servovalve spools and model channel (or equivalent) provide signals for comparison monitoring. Complete channel isolation is maintained. Failures are detected without requiring actuator motion from the command position.

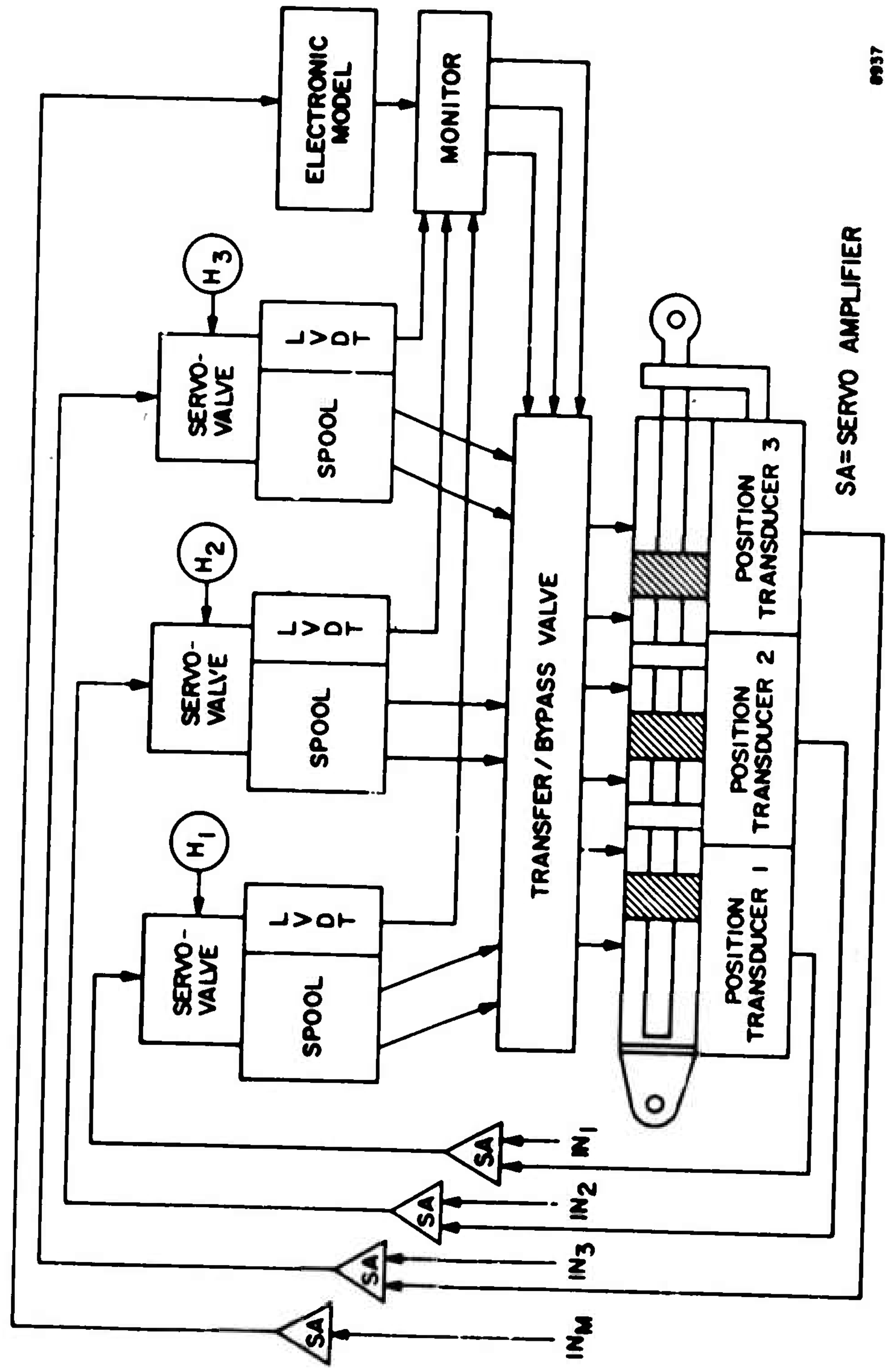


Figure 39
Model 1

For the first failure, the system switches from the active channel to a standby channel unless the failure is in a standby channel. In this case, the monitor prevents that channel from ever being engaged. For a second failure, the system switches to the remaining standby channel. In any operational mode, the engage valve bypasses the nonengaged actuators.

The model 1 configuration has the following advantages:

(1) No performance degradation due to failures since each channel can carry the load

(2) Failure isolation maintained

(3) Mechanization easily expanded to higher redundancy

(4) Eliminated mechanical linkages

(5) Actuator deviation is not required for failure detection

The configuration has the following disadvantages:

(1) Depends on solenoid valve and engage valve reliability for transfer

(2) Fast transfer times require high speed solenoids and comparators to minimize transients

(3) Monitor may be sensitive to large power transients

(4) Large size and weight because each actuator must be sized to carry the full load

(5) Monitor sensitive to large load variations because of the electronic model

(6) Spool position transducer reduces valve performance, increases cost, and lowers reliability

<u>Comments</u>:

This configuration could be designed for active redundancy using synchronization to eliminate disadvantages 2 and 4 but at the expense of added complexity.

b. Model 2. Spool-Monitored (Hydraulic) Standby Redundant Actuator

The model 2 configuration is very similar to model 1 except for the monitoring mechanization which is all hydraulic. Model 2 uses four real channels (identical within tolerances) with one acting as a model. Actuator position feedback is electrical. One channel is active while all others are in standby. The servovalves are coupled to an actuator through a four-position engage valve. The engage valve transfers the system through its operational modes on commands directly from the hydraulic monitor. The positions of the servovalve spools are measured and compared hydraulically. Complete channel isolation is maintained. Failures are detected without required actuator deviation from the commanded position.

For the first failure, the system switches from the active to a standby channel unless the failure is in a standby channel. In this case the monitor prevents that channel from ever being engaged. In any operational mode, the engage valve bypasses the nonengaged actuators.

The configuration has the following advantages:

(1) No performance degradation due to failures since each channel can carry the load

(2) Failure isolation maintained

(3) Monitor insensitive to load variations

(4) Mechanization easily expanded to higher redundancy

(5) Eliminates mechanical linkages

(6) Actuator deviation is not required for failure detection

(7) System transfer is very fast because electrohydraulic solenoids are not used

The configuration has the following disadvantages:

(1) Depends on comparator and engage valve reliability for transfer

(2) Hydraulic comparators are not fail-safe

(3) Increased size and weight because each actuator must be sized to carry the full load

(4) The close electrical tolerance required to match channels for small failure monitoring is costly

(5) Hydraulic logic is susceptible to silting effects.

c. Model 3. Secondary Actuator With Standby Redundancy

The model 3 configuration (figures 40 and 41) has a small redundant secondary servoactuator which mechanically drives the main control valve and

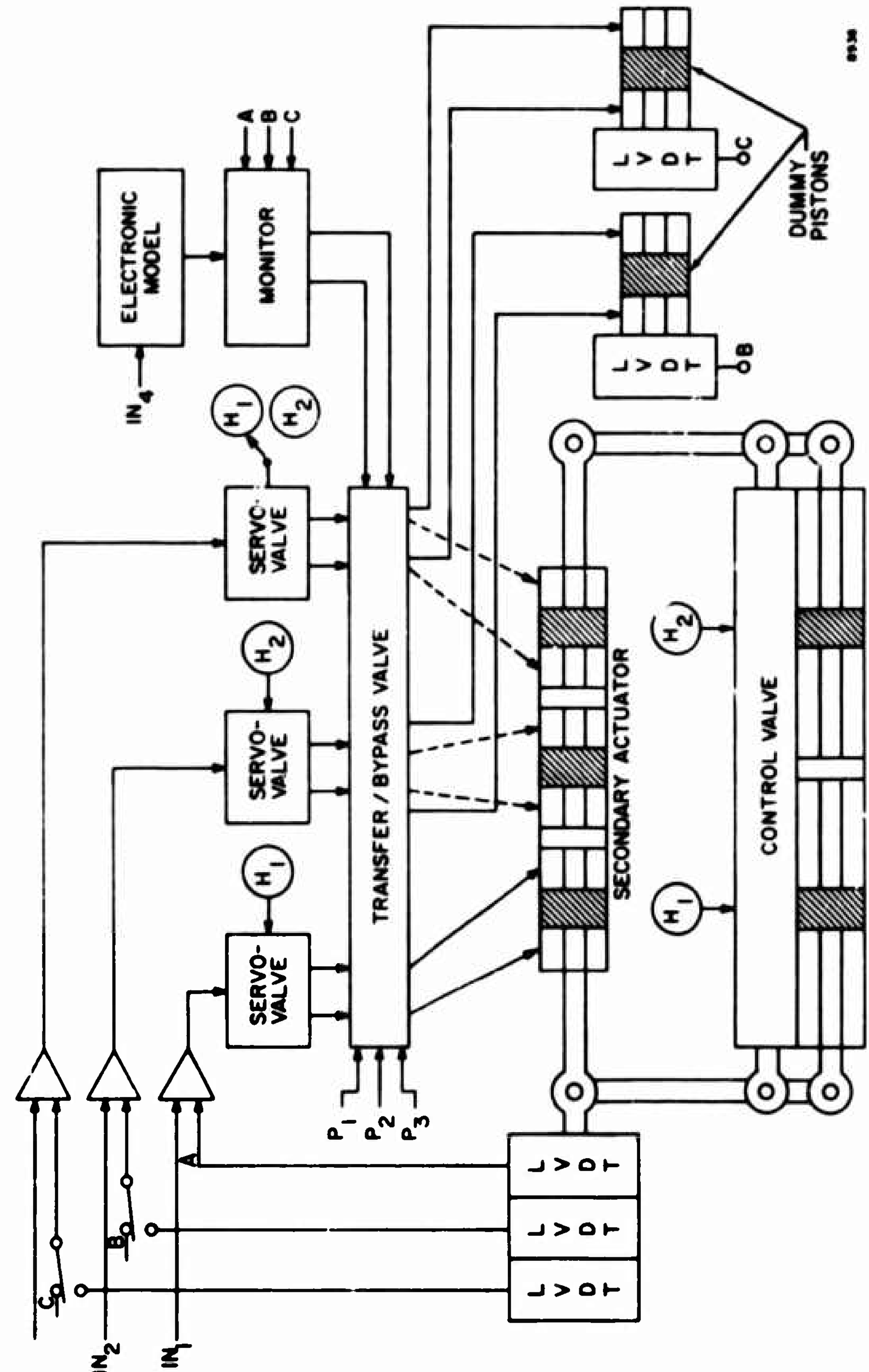


Figure 40
Model 3A

Comments:

This configuration could be designed for active redundancy using synchronization to eliminate disadvantages 2 and 4 but at the expense of added complexity.

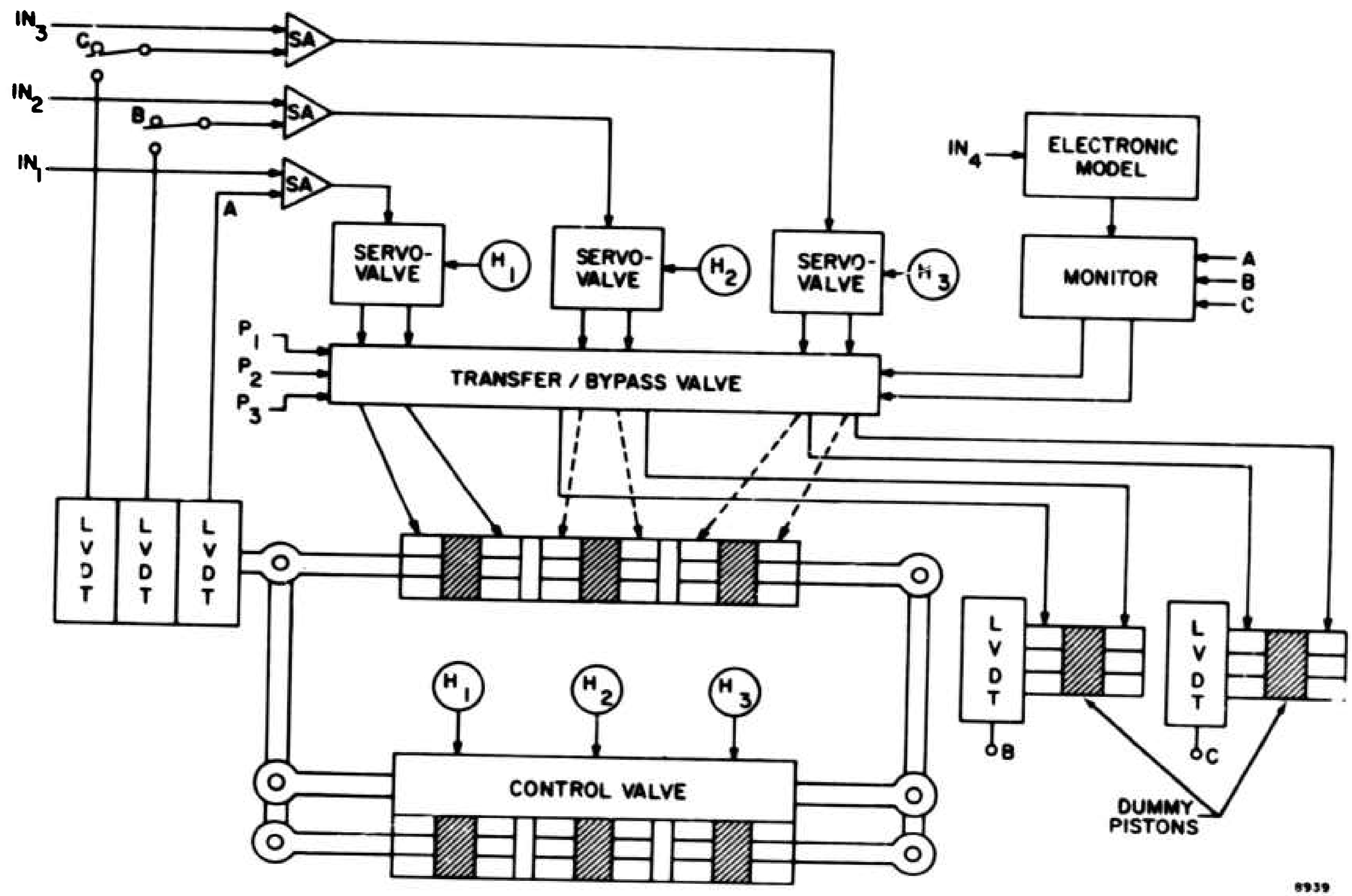


Figure 41
Model 3B

power actuator with nearly unity feedback. The dual feedback linkage can be sealed within the actuator body where it is protected and bathed in oil. The power actuator employs active redundancy. The secondary servo uses three real channels (identical within tolerances) and a model channel which may be hydraulic or electronic. One real channel is active while the others operate in standby by driving dummy model pistons which are sized to match the secondary actuator. The feedback of the model piston and the secondary actuator is electrical. The servovalves are coupled to the secondary actuator through a four-position engage valve. The engage valve transfers the system through its operational modes on commands from the electronic monitor via electrohydraulic solenoids. The monitor compares the position of the secondary actuator and model pistons. This eliminates the need for servovalve spool transducers and allows limited monitoring of the main control valve and power actuator. Complete channel isolation is maintained. Failures are detected without requiring power actuator motion from the commanded position although some motion will occur for hardover failures.

For the first failure, the system switches from the active secondary channel to a standby channel unless the failure is in a standby channel. In this case the monitor prevents that channel from ever being engaged. For the second failure, the system switches to the remaining standby channel. Channel switching proceeds as follows assuming an active channel failure. The engage valve disconnects the active valve from the actuator and bypasses the actuator while simultaneously switching the standby valve from its model piston to the secondary actuator. Also simultaneously, the monitor switches the feedback of the standby channel from its model piston to the standby feedback transducer on the secondary actuator.

The configuration has the following advantages:

(1) No performance degradation due to failures

(2) Failure isolation is maintained (Model 3B only)

(3) Monitor insensitive to load variations

(4) Mechanization easily expanded to higher redundancy

(5) Actuator deviation not required for failure detection

(6) Servovalve spool position transducers not required

(7) Small servovalves are adequate

(8) Power servo channels are active which minimizes size

The configuration has the following disadvantages:

(1) Depends on solenoid valve and engage valve reliability for transfer

(2) Fast transfer times require high speed solenoids and comparators to minimize transients

(3) Monitor may be sensitive to large power transients

(4) Part of secondary failure transient is transmitted to the output through the mechanical linkage

(5) Requires a secondary actuator

(6) Requires model pistons and/or an electronic model

d. Model 4. Fail-Passive Secondary Actuator

The model 4 configuration (figures 42 and 43) is described in somewhat more detail because of its unique characteristics. It employs a small redundant secondary actuator which mechanically drives the main control valve and power actuator with nearly unity feedback (similar to model 3). The dual mechanical linkage can be sealed within the actuator body where it is protected and bathed in oil. Both the secondary and power actuators employ active redundancy. When dual hydraulic supplies are used, the secondary actuator is dual tandem with two single-stage jet-pipe valves driving each piston thus forming four inner servo loops. When triple hydraulic supplies are used, the secondary actuator is triple tandem with a single valve driving each piston thus forming three inner loops.

The uniqueness of the configuration derives from the inner loops which are designed to have passive failure characteristics. A fail-passive channel fails in such a way that it has no output and it does not interface with the normal operation of a parallel channel. In other words, active or hardover failures have been eliminated by design. Since a failed channel has no force output, the other good channels can operate unimpeded. The single-stage jet-pipe valve not only has the proper failure characteristics, but it also acts like a very open-centered valve so that fluid can be forced back through it with relative ease thus preventing hydraulic lock. The servo error signal is formed in the position feedback transducer, rather than in an amplifier as is normally done, such that a transducer failure blocks the command signal. This feature prevents the open loop condition that normally results from a loss of feedback. The electronics fail passively because ac signals are used. A hardover electronic failure causes a dc output to which the ac circuits are not sensitive.

If a hardover input should occur in a channel or as an input, the other channels collectively offset the output force of the failed channel at the force-summing actuators. The high loop gains reduce the resulting position offset to an insignificant level. Therefore, a quadruplex servo with four hydraulic and electrical supplies will operate after three failures, and a triplex servo will operate after two failures. Further, no monitoring, switching, or engage valves are required in this approach. Monitoring is performed, however, primarily for failure reporting. In the triplex servo, a hardover monitor may be used to provide center and lock, in the event that one of the three failures is not passive.

Both the fail-passive triplex and quadruplex servos have been tested in the laboratory to demonstrate their operation and performance. The quadruplex servo operates slightly better than predicted by theory. This is because a failed servo does not completely bypass the other channel on the

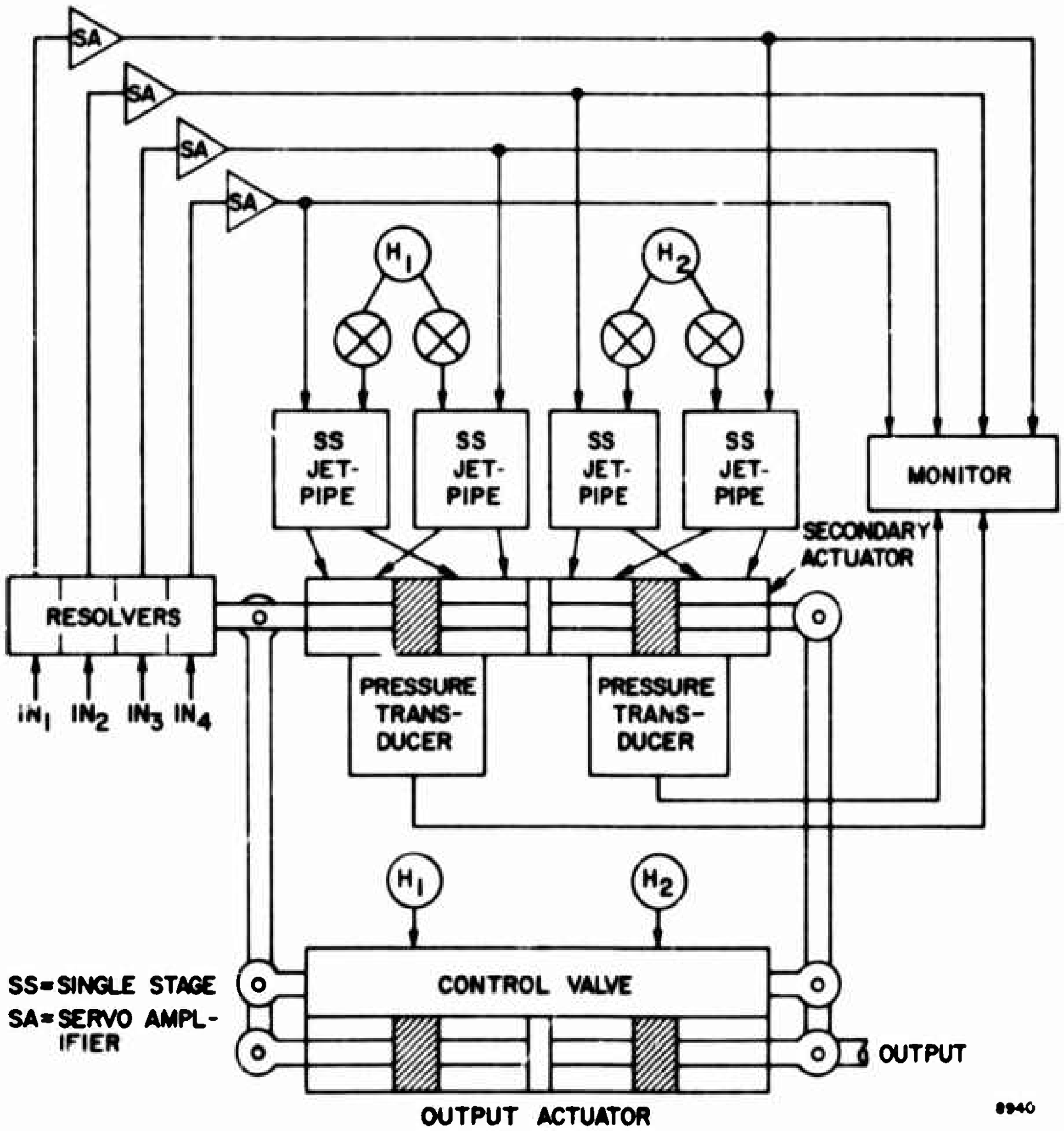


Figure 42
Model 4A

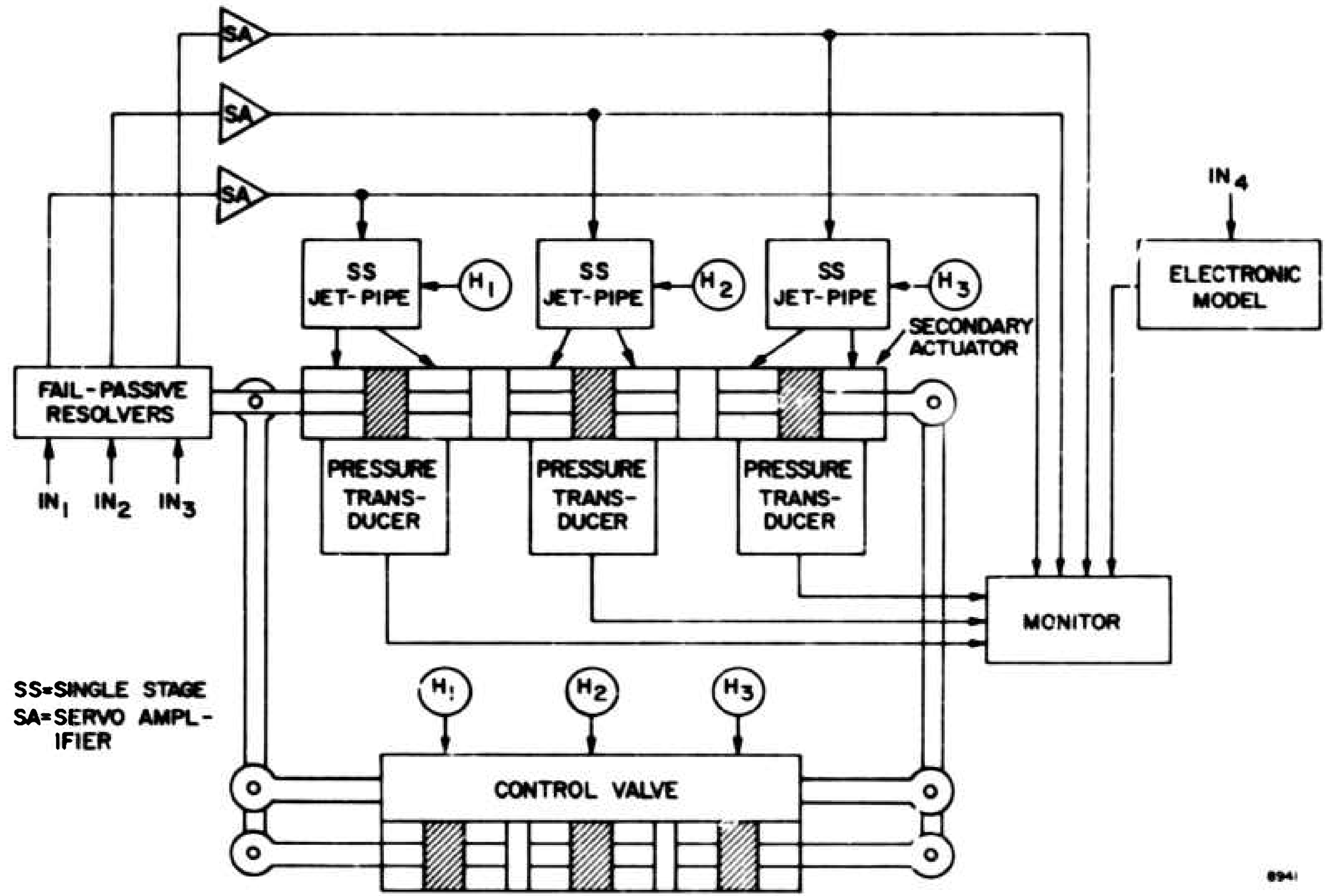


Figure 43
Model 4B

same piston. Therefore, after three failures, the actuator retains about 20 percent of its dynamic performance which would likely be enough to let the pilot fly the aircraft. An important point to note is that the servo operates after three failures without the use of monitoring or switching. The lack of switching not only simplifies the design but it also eliminates the failure transient problem. Relatively simple monitoring utilizes differential pressure transducers to measure the output force of the servos. The monitor correlates this information with the error signals to determine which channels have failed. A final point is that the fail-passive servo does not require tight tolerances because it need not be monitored. On the experimental model, the tolerances were purposely varied by ±30 percent with no noticeable effect on performance. This result has obvious advantages in the economy of construction and operation.

The configuration has the following advantages:

(1) No switching required for failures thus eliminating switching transients and engage valves

(2) Relatively simple monitoring required for failure reporting only

(3) A triplex system remains operational after two failures; a quadruplex system remains operational after three failures; etc

(4) Size, weight, complexity and cost are minimum for the given degree of redundancy

(5) Very tolerant to channel mismatches

(6) Easily adapted to any degree of redundancy

(7) Requires single-stage valves rather than two-stage valves which improves reliability

(8) Very tolerant to dirty fluid; can operate with 200 micron filters

The configuration has the following disadvantages:

(1) Requires a secondary actuator

(2) May have limited dynamic performance and threshold in very high performance applications when using presently available single-stage jet-pipe valves

(3) The triplex configuration requires an electronic model to ensure center and lock for a third failure

(4) Force degradation for hydraulic failures

e. Model 5. Position-Monitored Standby Redundant Actuator

The model 5 configuration employs three parallel position servo-actuators with common outputs and an electronic model and monitoring (figure 44). Each servo channel consists of an actuator, two-stage servo-valve, servo amplifier, and dual LVDT position transducers. The actuators are tied together rigidly so that relative motion does not occur. The outer channel cylinder bodies are connected through a differential link pivoted at the center where it is attached to the main actuator body. The outer channels are normally active. The center actuator is normally in standby; it engages only if both outer channels fail. This is a hybrid active/standby type of redundancy.

Operation of the outer channels (A and B) is normally like a position summed actuator. The differential link motion is small (caused by tolerance variations) if the A and B outputs are equal. Output motion equals the average output of A and B. On failure of either A or B, the failed actuator is bypassed and its link end is locked. The outer channel then supplies the output with undegraded performance. Failure of the second channel causes its actuator to be bypassed, its end link to be locked, and the center actuator (C) to be engaged. Channel C then drives the load with undegraded performance. A third failure centers and locks the actuator. The locks, bypass valves, and engage valves operate on command from the electronic monitor via electrohydraulic solenoids. The monitor operates on the actuator position transducer outputs and the model output.

The configuration has the following advantages:

(1) No performance degradation after failures

(2) Failure isolation is maintained

(3) Mechanization expandable to higher redundancy

(4) Servovalve spool position transducers not required

The configuration has the following disadvantages:

(1) Output deviation required for failure detection

(2) Depends on solenoid valve, bypass valve, and lock and monitor reliability for transfer

(3) Fast transfer times require high speed solenoids and monitor to minimize transients

(4) Monitor sensitive to large load variations because of the standby channels

(5) Monitoring may be sensitive to large power transients

(6) Increased size and weight because each actuator must be sized to carry the load

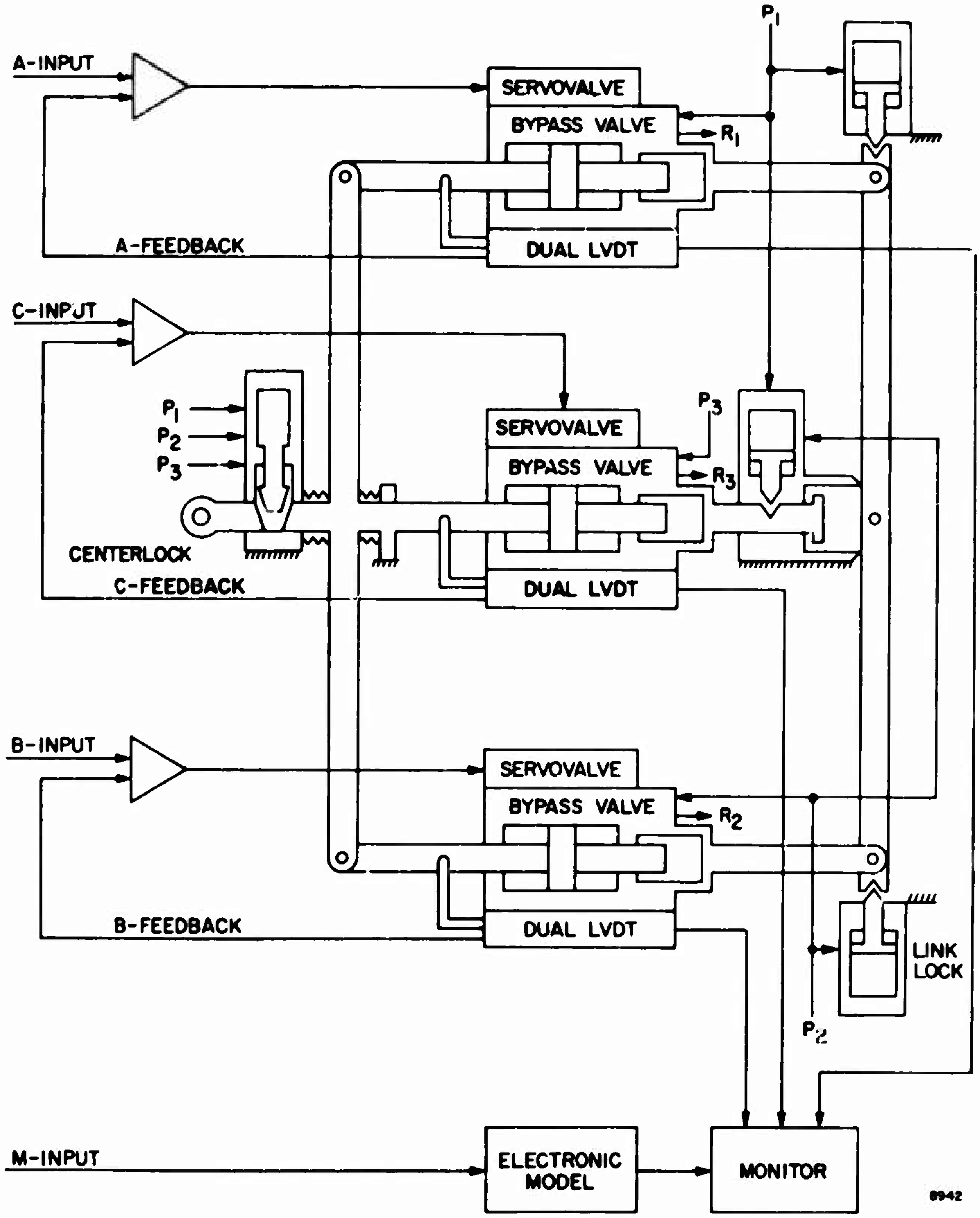


Figure 44
Model 5

Comments:

By using the output as a small secondary actuator to mechanically drive a power actuator valve, disadvantages 1, 4, and 6 could be minimized or eliminated.

f. Model 6. Position-Monitored (Hydraulic) Standby Redundant Actuator

The model 6 configuration shown in figure 45 employs four parallel position servoactuators whose outputs are rigidly connected so that relative motion does not occur. One channel (A) is active, two channels (B and C) are in standby, and one channel (D) is a model. A channel consists of an actuator, two-stage servovalve, servo amplifier, and LVDT position transducer. A series of engage valves and locks connect the various channels to the load, one at a time, on command from the hydraulic monitor. The actuator cylinders are sleeves that move within the main actuator body against a centering spring load except channel A which is the reference channel whose sleeve is fixed. Porting between the sleeves and the body provides hydraulic position comparison with the reference channel for voting. Under normal conditions no relative motion occurs. Upon failure of channel A, relative motion occurs in all three sleeves. Channel A is bypassed and the sleeve of channel B is locked to the body thus engaging it to the load and making it the new reference channel. If any other channel fails first, its sleeve alone moves. The resulting vote causes the actuator of that channel to be bypassed. A third failure causes center lock because of the disagreement of sleeve positions.

The configuration has the following advantages:

(1) No performance degradation after failures

(2) Failure isolation maintained

(3) Mechanization easily expanded to higher redundancy

(4) Servovalve spool transducers not required

(5) Transfer is fast because electrohydraulic solenoids are not used

The configuration has the following disadvantages:

(1) Output deviation required for failure detection

(2) Depends on engage valve and lock reliability for transfer

(3) Silting may affect hydraulic comparator performance by increasing threshold

(4) Increased size and weight because each actuator must be sized to carry the load

(5) Monitor sensitive to large load variations because of the standby channels

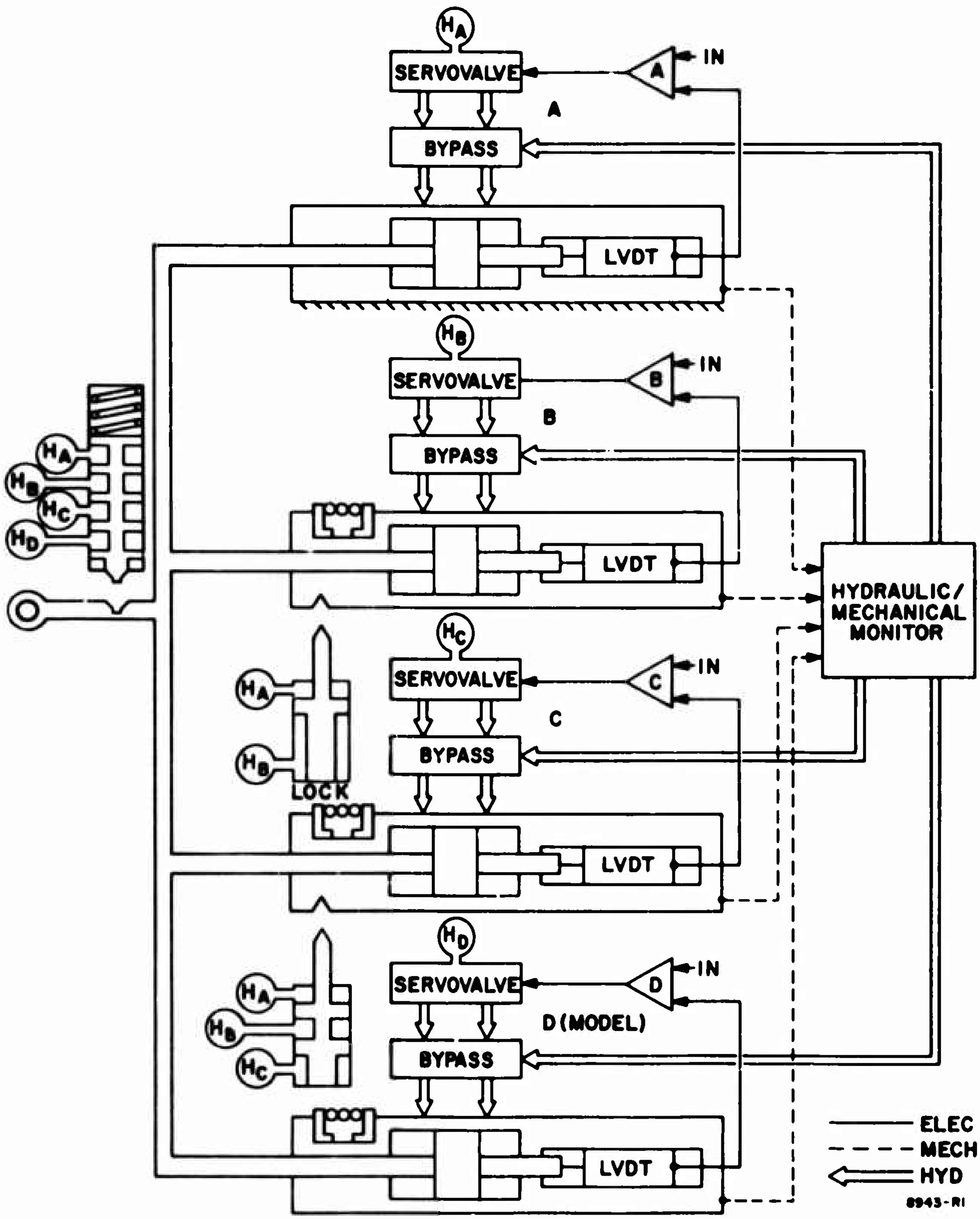


Figure 45
Model 6

Comments:

By using the output as a small secondary actuator to mechanically drive a power actuator valve, disadvantages 1, 4, and 5 can be minimized or eliminated.

g. Model 7. Force-Summed Voted Actuator

The model 7 configuration, shown in figure 46, has been developed and tested by Elliott Brothers of England. It employs four separately controlled hydraulic actuators coupled in parallel to a common output member by means of miniature hydraulic couplings combined with ball clutches. Each hydraulic coupling has a zero rate springbox characteristic, and its stroke is determined by the tolerance between channels necessary to allow for component variations. If failure causes the coupling to reach the end of its stroke, the balls disengage from a groove in the common member and so declutch the failed actuator from the common output. The clutch mechanism is a variation of the well-known quick-release self-sealing hydraulic coupling which is in widespread use.

A simple gate mechanism is provided to prevent more than two channels from becoming disengaged at any one time. This gate is required to prevent disengagement of more than two channels which might otherwise occur due to some remote common cause, such as an excessive output load on the actuator, and cause a loss of control. The pilot can be warned of a declutched actuator channel by means of a failure display panel. The actuator remains disconnected from the common output until the clutch is re-engaged. This is effected electrically by means of remotely operated solenoids which are operated from the cockpit. The real value of the remote re-engagement facility is to allow complete checking of separate control channels without the need for complex test equipment.

Each actuator has electrical feedback. In addition a low-gain mechanical feedback centers the actuator in the event of the loss of electrical power. The actuator centers automatically when either hydraulic supply is on, independent of electrical power. This action is equivalent to mechanical spring centering which is the conventional but heavier method. The mechanical feedback applies enough force to the flapper of the servovalve to cause the actuator to return to the midposition. The gain of the mechanical feedback is such that the performance of the actuator is dominated by the electrical feedback loop.

The configuration has the following advantages:

(1) Failure isolation is maintained

(2) Mechanization flexible with respect to redundancy

(3) Servovalve spool transducers not required

(4) Channel transfer is very fast because electrohydraulic solenoids are not used

The configuration has the following disadvantages:

(1) Output deviation (although very small) required for failure detection

(2) Force-voting mechanism will tend to get large and heavy for high power actuators

(3) Depends on voting mechanism reliability for transfer

Comments:

By employing the output member as a secondary actuator, disadvantages 1 and 2 would be eliminated.

6. CANDIDATE SYSTEMS

The candidate fly-by-wire systems which have evolved from analysis thus far can be reasonably well defined with the exception of the actuators. These require further evaluation which will be accomplished through simulation techniques as will be described in the next section. Except for the actuator configuration then, the system takes on the form generally expressed in figures 29, 30, and 31 and includes the following equipment:

a. Spring-centered control stick having ac position transducers

b. Mixed ac and dc electronics, for signal shaping, summing, and monitoring, using both microcircuits and discrete components packaged in potted modules by channel

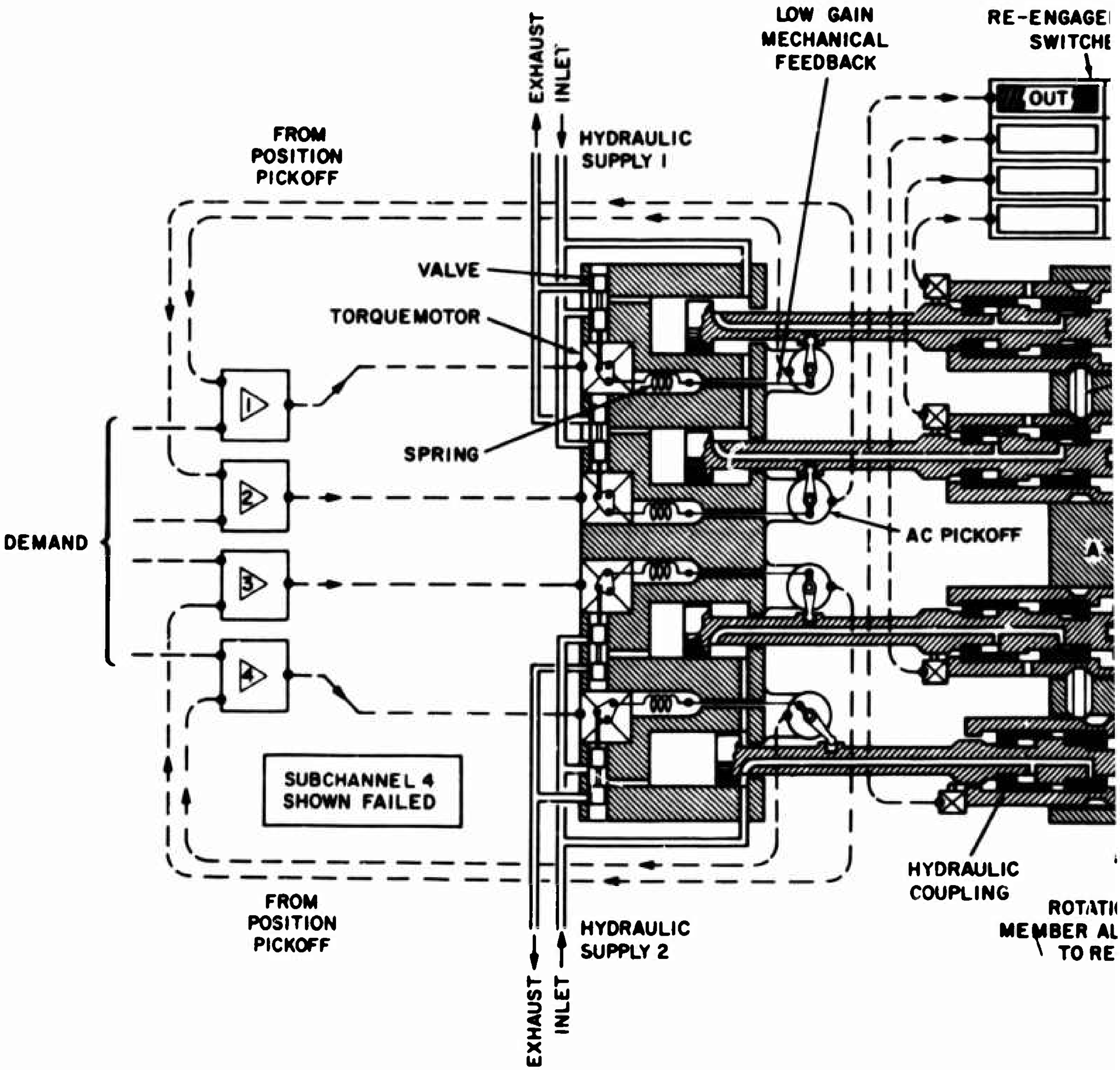
c. C* feedback utilizing rate gyros and direct-measuring normal accelerometers incorporating self-test capability

d. Series trim (preferred)

e. Reinforced copper wire transmission line protected by conduit where necessary

f. Triplex or quadruplex redundancy

g. Monitor at the actuator just before its output and at the C* sensors just before summing with the command signal


LOW GAIN
MECHANICAL
FEEDBACK
RE-ENGAGE
SWITCHE
OUT
EXHAUST
INLET
HYDRAULIC
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VALVE
TORQUEMOTOR
SPRING
DEMAND
1
2
3
4
AC PICKOFF
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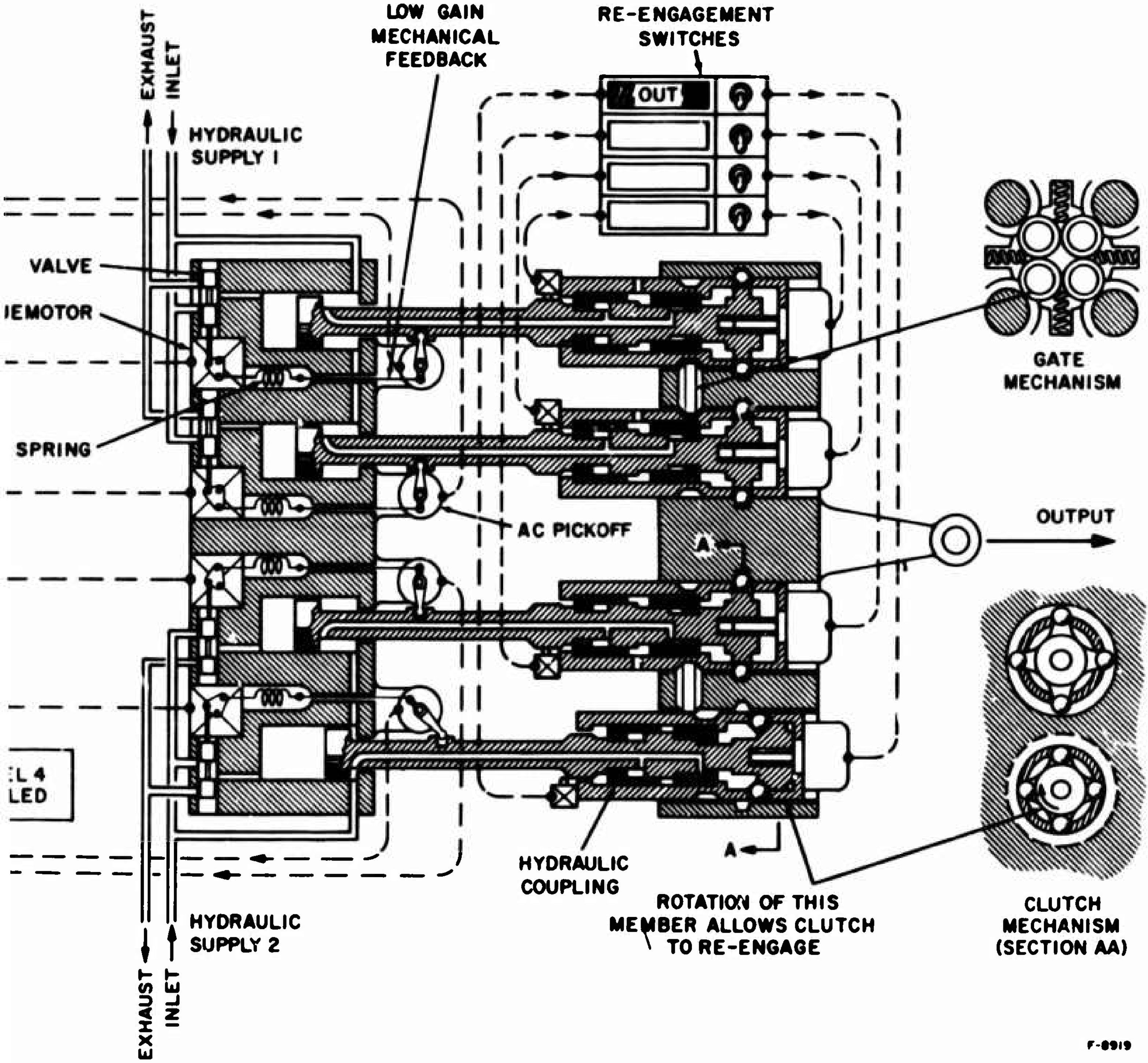


Figure 46
Model 7

12

While the actuator configuration analysis does not contain sufficient information for an accurate comparison of reliability, cost, weight, and volume, the comparison of 13 relevant factors summarized in table V provides some interesting answers. Relative reliability can be gaged by factors 4, 9, 10, and 12. The estimated number of moving parts provides a clue not only to the relative reliability but to weight and volume as well. Factors 9 and 10 relate to the ability of a configuration to remove a failed channel which also greatly affects reliability. Factors 8 and 11 are important to the relative cost. In particular tightly controlled tolerances are expensive both in parts cost and in assembly. Factor 1 shows that all of the configurations can get down to a 30-millisecond transfer time with a little effort. However, several configurations beat this time with no effort. The hydraulic fluid filtering requirement, factor 12, is another often overlooked design factor. The requirements range from the 3-micron filter on the hydraulic comparators of Hydraulic Research's hydrologic through the typical 10-micron filters to the 50-micron (plus) requirement for Sperry Phoenix's fail-passive actuator. The argument for fine filtration is a good one in that very clean fluids cause very little if any silting, plugging, or jamming effects. This obviously improves reliability. However, on the other hand, the finer the filter, the more often it must be replaced.

A 20-micron system filter is replaced about every 1,000 hours, and a 10-micron filter is replaced every 100 hours. Continuing this progression would show that a 5-, 2-, and 1-micron filter would require replacement approximately every 10, 1, and 0.1 hours respectively. This obviously shows why 5-micron and smaller system filters are seldom used. Furthermore, while 5- and 10-micron particles can be removed from the system, 1- and 2-micron particles cannot because they are continuously being generated within the system (even if it were sealed) from the seals, wear, and other nondescript sources. Because of the very large population of such particles, even filters which do not have large flows, such as in the first stage of the servovalve, can quickly collect a large quantity of particles and become clogged. A system that can operate in a dirty environment does not have the replacement problem.

The reasons for selecting the fail-passive approach over the others for use in fly-by-wire systems is obvious from the table. It has no failure transfer time, little degradation, the fewest number of moving parts (hence by implication the most reliable, lightest, and smallest), it is the only configuration that does not depend on nonredundant components for switching out failures (such as engage valves, solenoids, or monitors), it is easily the most tolerant of channel mismatch (which implies low cost and a very low nuisance failure indication rate), and it is also the most tolerant of dirty hydraulic fluid.

The second and third choices of actuator configuration are models 2 and 3 respectively. Model 2 has a very fast switch time which is important in minimizing airframe transients. However, the tight tolerances required may be expensive to obtain and expensive to maintain as well since parameter drift tolerances must also be small. A tradeoff should be possible between longer larger failure transients and larger tolerance requirements. Model 3

would be equally acceptable as model 2 if the minimum transfer time were achieved. This would require limited solenoid development to provide faster switching. The airframe response to various switching times will be demonstrated in the next section.

The list of actuator models is repeated to aid in using table V.

a.	Model 1	Conventional standby-redundant actuator with electrical monitoring on valve spool position.
b.	Model 2	Conventional standby-redundant actuator with hydraulic monitoring on valve spool position.
c.	Model 3	Secondary actuator with standby-redundancy.
d.	Model 4	Fail-passive secondary actuator.
e.	Model 5	Standby-redundant actuator with electrical position monitoring.
f.	Model 6	Standby-redundant actuator with hydraulic position monitoring.
g.	Model 7	Force-summed voted actuator.

TABLE V

SUMMARY COMPARISON OF ACTUATOR CONFIGURATIONS

Factor	Actuator Configuration Model 1	2	3	4	5	6	7
1. Time to detect and switch out failures (milliseconds)	30-100	4-10	30-100	0	30-100	10	10
2. Performance degradation after one failure	No	No	No	No	No	No	25%
3. Performance degradation after two failures	No	No	No	10%	No	No	50%
4. Mechanical complexity: Number of moving parts	18	30	22	11	22	18	33
5. Isolated failures	Yes	Yes	Yes	Yes	Yes	Yes	Yes
6. Mechanically flexible to redundancy implementation	Yes	Yes	Yes	Yes	Yes	Yes	Yes
7. Output deviation required for failure detection	No	No	No	No	Yes	Yes	Little
8. Requires servovalve spool position transducer	Yes	Yes	No	No	No	No	No
9. Maximum allowed tolerance buildup (percent channel mismatch)	±20	±10	±20	±50	±20	±10	±20
10. Hydraulic fluid filter required (microns)	10	2-5	10	50+	10	2-5	10
11. Monitor sensitive to: Load	No	No	No	No	Yes	Yes	Yes
Power variations	Yes	No	Yes	No	Yes	Yes	No
Dirt	No	Yes	No	No	No	Yes	No

SECTION VII

SIMULATION STUDY

1. BREADBOARD MODEL

The simulation portion of the fly-by-wire program was done primarily to evaluate the actuator configurations discussed in Section VI and to study C* command responses. By means of analog simulation, with breadboard hardware where feasible, the intention was to verify operation, to discover possible hidden problems, and to evaluate the candidate systems with respect to failure induced transients and channel transfer times.

The block diagrams in figures 47, 48 and 49 show the model 3, 4, and 2 candidate systems respectively and their transfer equations. An investigation of the model and systems reveals that they differ only in the dynamic performance of the secondary actuator/single-stage jet-pipe valve combination, figure 47, and the two-stage valve of figure 49. While this is not an obvious result when comparing the block diagrams of figures 47 and 49, the following transfer functions are produced:

From figure 47

$$\frac{\delta_1}{C^*_\epsilon} = \frac{(100)\left(\frac{1}{80}\right)\left(\frac{4.55}{S}\right)}{1 + (100)\left(\frac{1}{80}\right)\left(\frac{4.55}{S}\right)(10)} \tag{12}$$

$$\frac{\delta_1}{C^*_\epsilon} = \frac{5.6875G}{S + 56.875} \tag{13}$$

$$\frac{\delta_e}{\delta_1} = \frac{400}{S^2 + 20S + 400} \tag{14}$$

$$\frac{\delta_e}{C^*_\epsilon} = \frac{(5.6875)(400)G}{(S + 56.875)(S^2 + 20S + 400)} \tag{15}$$

From figure 49

$$\frac{\delta_e}{C^*_\epsilon} = \frac{(10)\left(\frac{500}{S+500}\right)\left(\frac{20}{S(S+20)}\right)(G)}{1 + (10)\left(\frac{500}{S+500}\right)\left(\frac{20}{S(S+20)}\right)(2)} \tag{16}$$

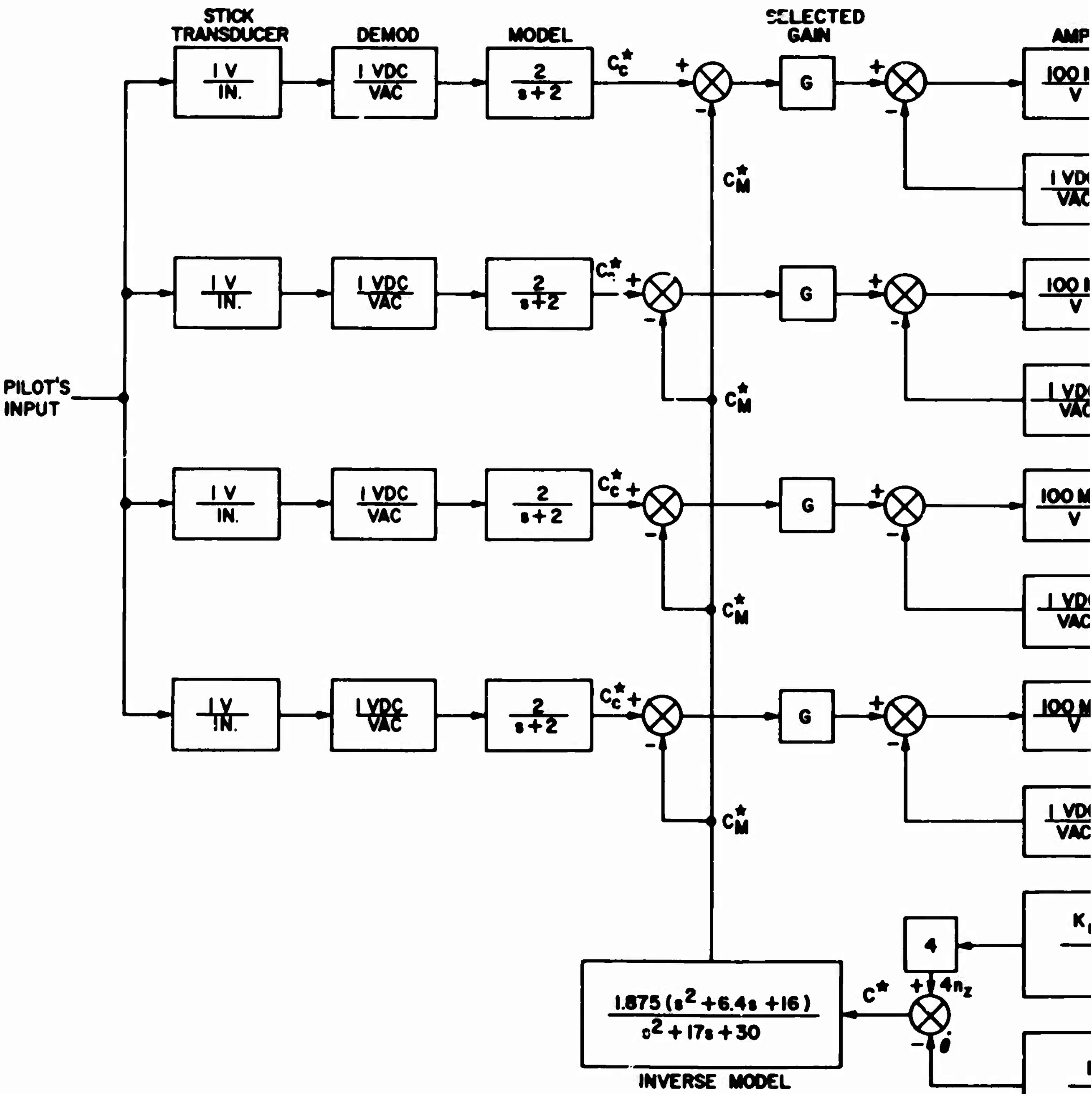
$$\frac{\delta_e}{C^*_\epsilon} = \frac{1000000G}{S^3 + 520\,S^2 + 10{,}000S + 200{,}000} \tag{17}$$

$$\frac{\delta_e}{C^*_\epsilon} = \frac{250}{(S + 500)}\;\frac{(400)G}{(S^2 + 19.2\;S + 400)} \tag{18}$$

A comparison of equations (15) and (18) shows that differences occur in the first order breaks (56.875 versus 500 radians) and in the gradient. It was therefore decided to simulate model 2 through the secondary actuator approach by increasing the break frequency by a factor of 10 in the secondary actuator simulation. No significant change in aircraft transient response occurred.

Since the majority of the simulation work to analyze and evaluate candidate systems involves the actuators, it was decided to simulate as much of the system as possible by analog computer. This results in better flexibility, and breadboard hardware is easier to add as it becomes available. The fail-safe comparators, logic, and channel selection switches were implemented by breadboard hardware. Figure 50 shows these circuits as they were breadboarded for the simulation. In the background is the analog patchboard. Figure 51 is a circuit diagram for the fail-safe comparator (self-indicating) used in the monitor circuits.

The comparator compares the difference between two channel error signals against a reference and produces an output only when it has not tripped. The action of the comparator is obtained by employing an ac excitation signal in addition to the difference signal to the input of operation amplifier A_1 of figure 51. The output signal of A_1 is then the sum of the ac excitation and the difference voltage. This signal is rectified to produce the bias voltage necessary to keep T_1 and T_2 conducting and the input to the logic circuit at ground. Inputs which saturate A_1, failure within the comparator, or loss of ac excitation results in the loss of the bias voltage to T_1 and T_2, and therefore the logic goes high and the comparator indicator goes off. Latching capability for the comparator is produced by positive feedback through the zener diodes, D_1. C_1 is adjusted to obtain the variable channel selection times of 10, 25, and 50 milliseconds. Due to time scaling in the analog computer section, the switch time is increased by a factor of 10 for compatibility. Figure 52 shows a circuit diagram for the logic and switching circuits necessary to obtain two-fail-operational capability. The logic elements are diode "hand" gates with transistor outputs. The logic circuit, designed with flatpack microcircuits, receives its inputs from the monitor comparator and provides drive signals for the channel selection switches.


STICK TRANSDUCER
DEMOD
MODEL
SELECTED GAIN
AMP
1 V / IN.
1 VDC / VAC
2 / s+2
C_c^*
C_M^*
G
PILOT'S INPUT
$\frac{1.875(s^2+6.4s+16)}{s^2+17s+30}$
INVERSE MODEL
C^*
4
$4n_z$
$\dot{\theta}$

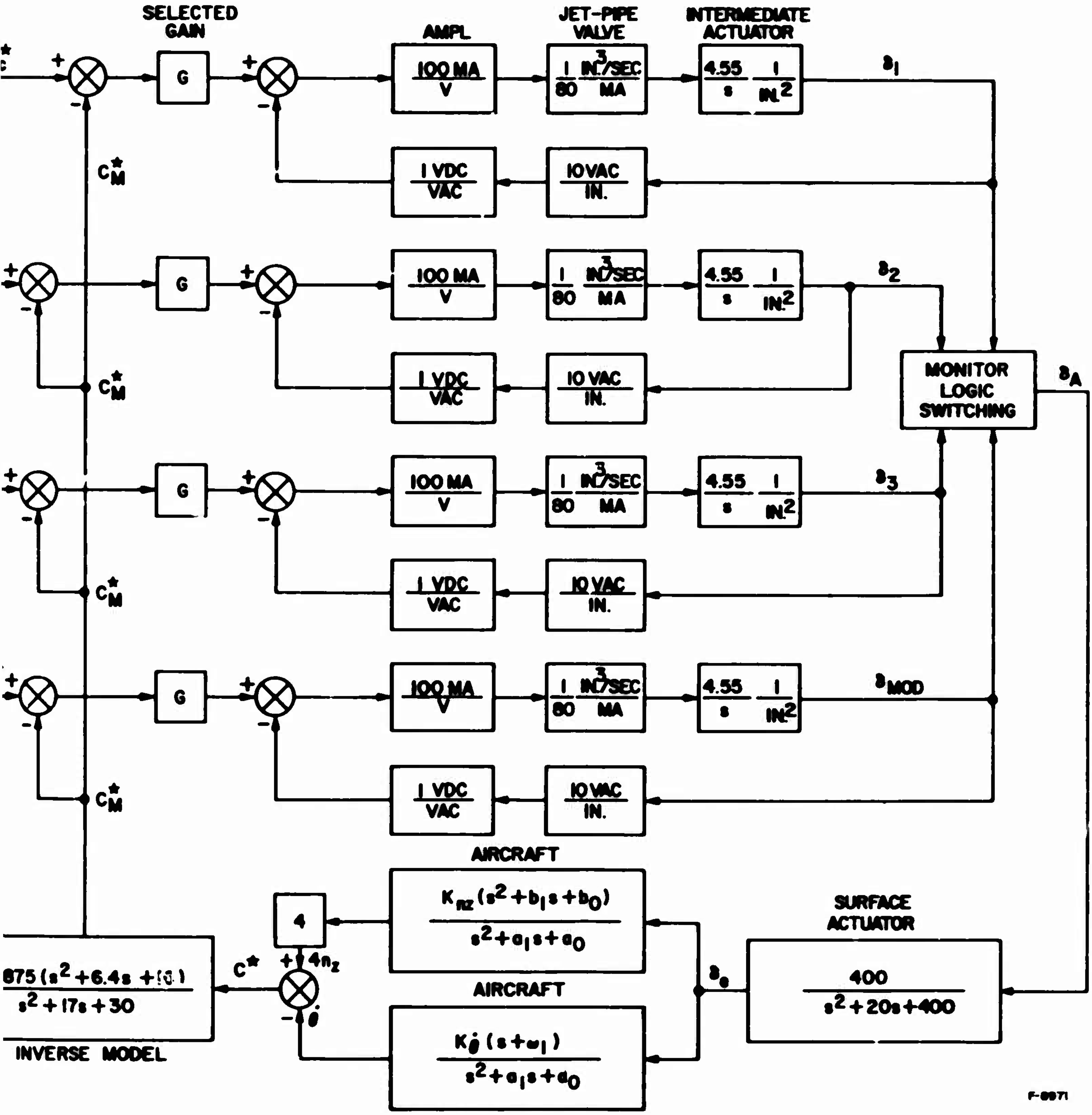
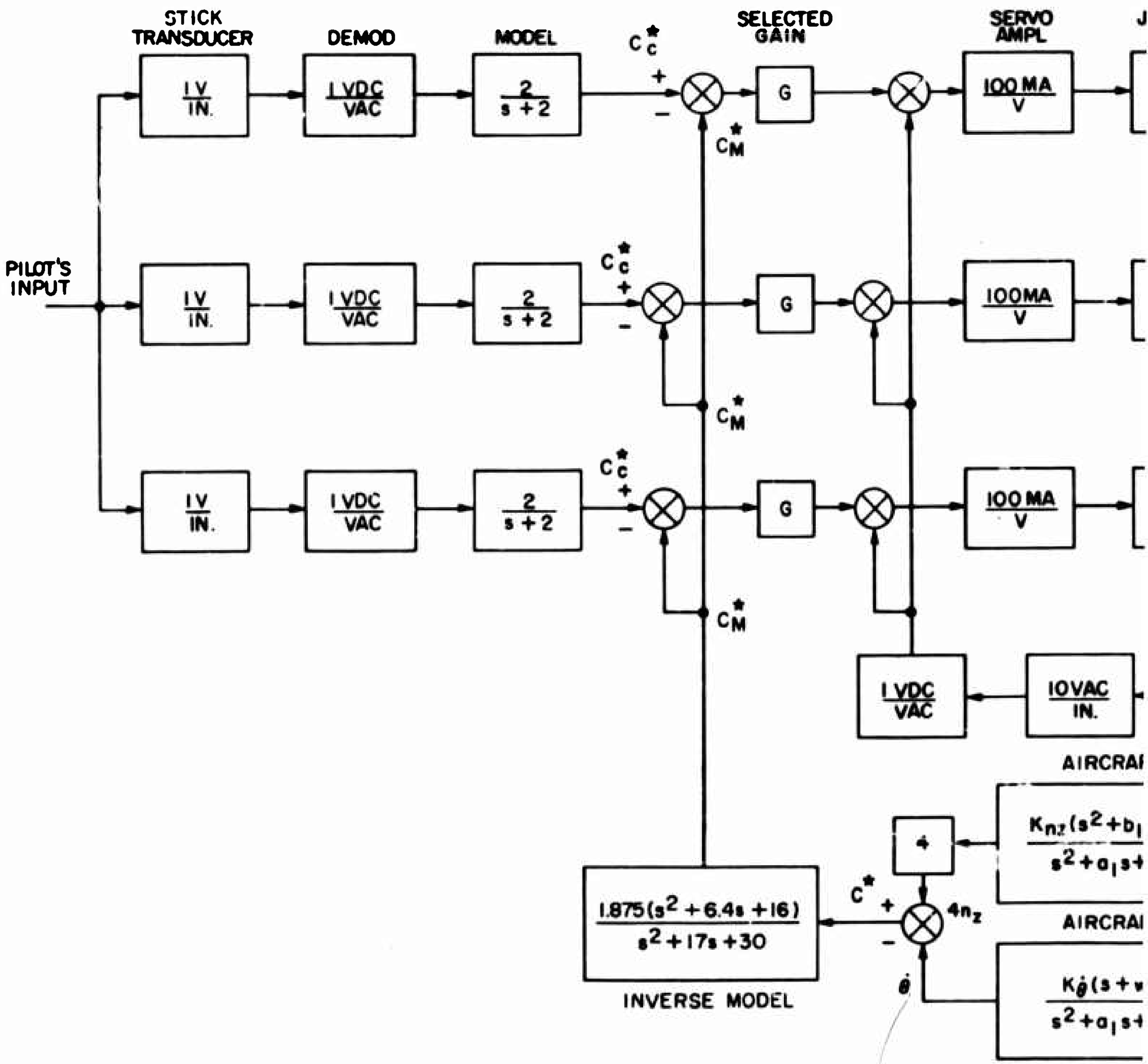


Figure 47
Block Diagram for Secondary
Actuator Concept

12


STICK TRANSDUCER
DEMOD
MODEL
SELECTED GAIN
SERVO AMPL
PILOT'S INPUT
1 V / IN.
1 VDC / VAC
2 / s + 2
G
100 MA / V
C_C^*
C_M^*
1 VDC / VAC
10 VAC / IN.
AIRCRAF
AIRCRAI
4
$4n_z$
C^*
$\dot\theta$
$1.875(s^2 + 6.4s + 16) / s^2 + 17s + 30$
INVERSE MODEL
$K_{n_z}(s^2+b_1$
s^2+a_1s+
$K_{\dot\theta}(s+$
s^2+a_1s+

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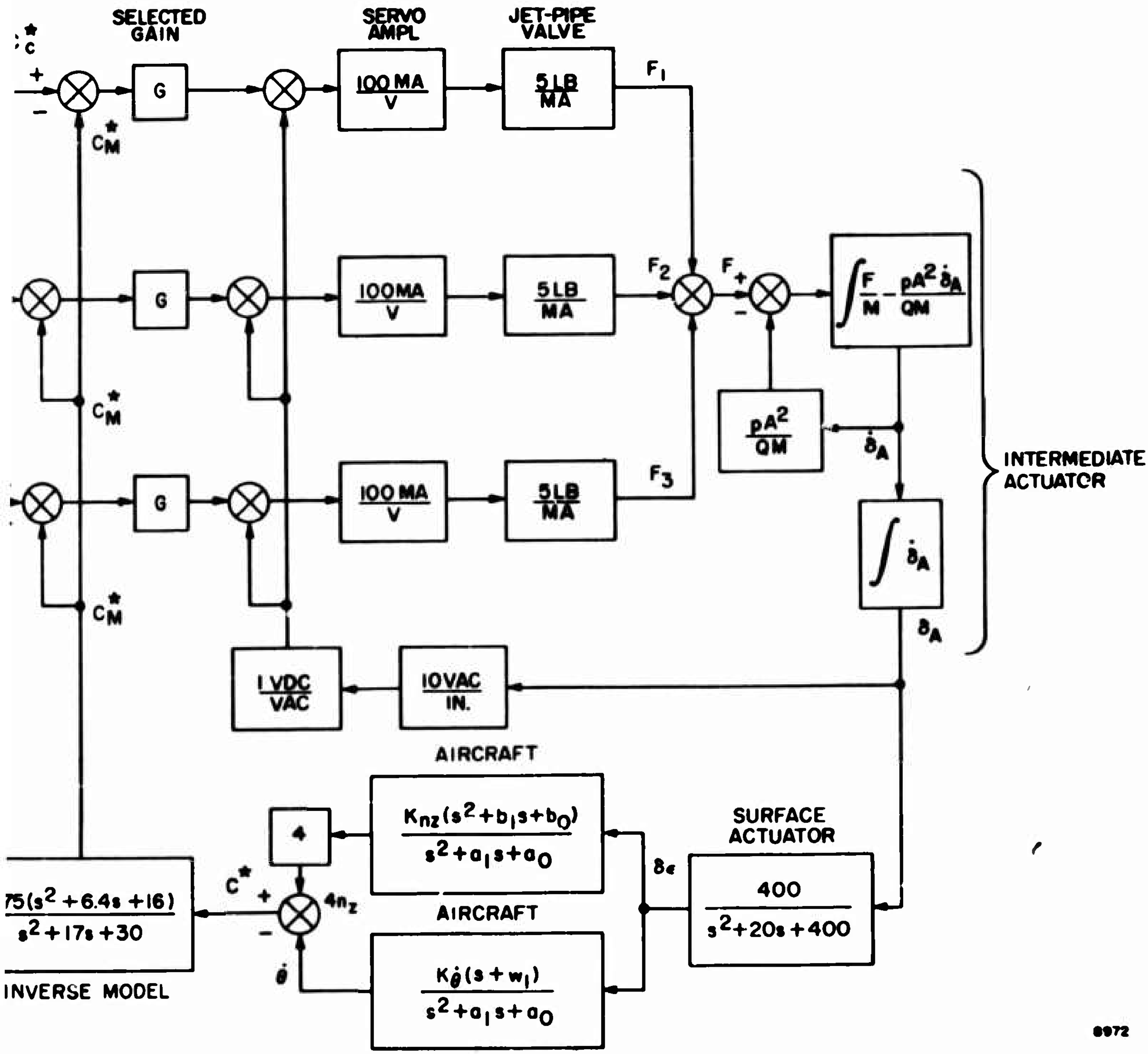

Figure 48
Block Diagram for Fail-Passive Concept

12

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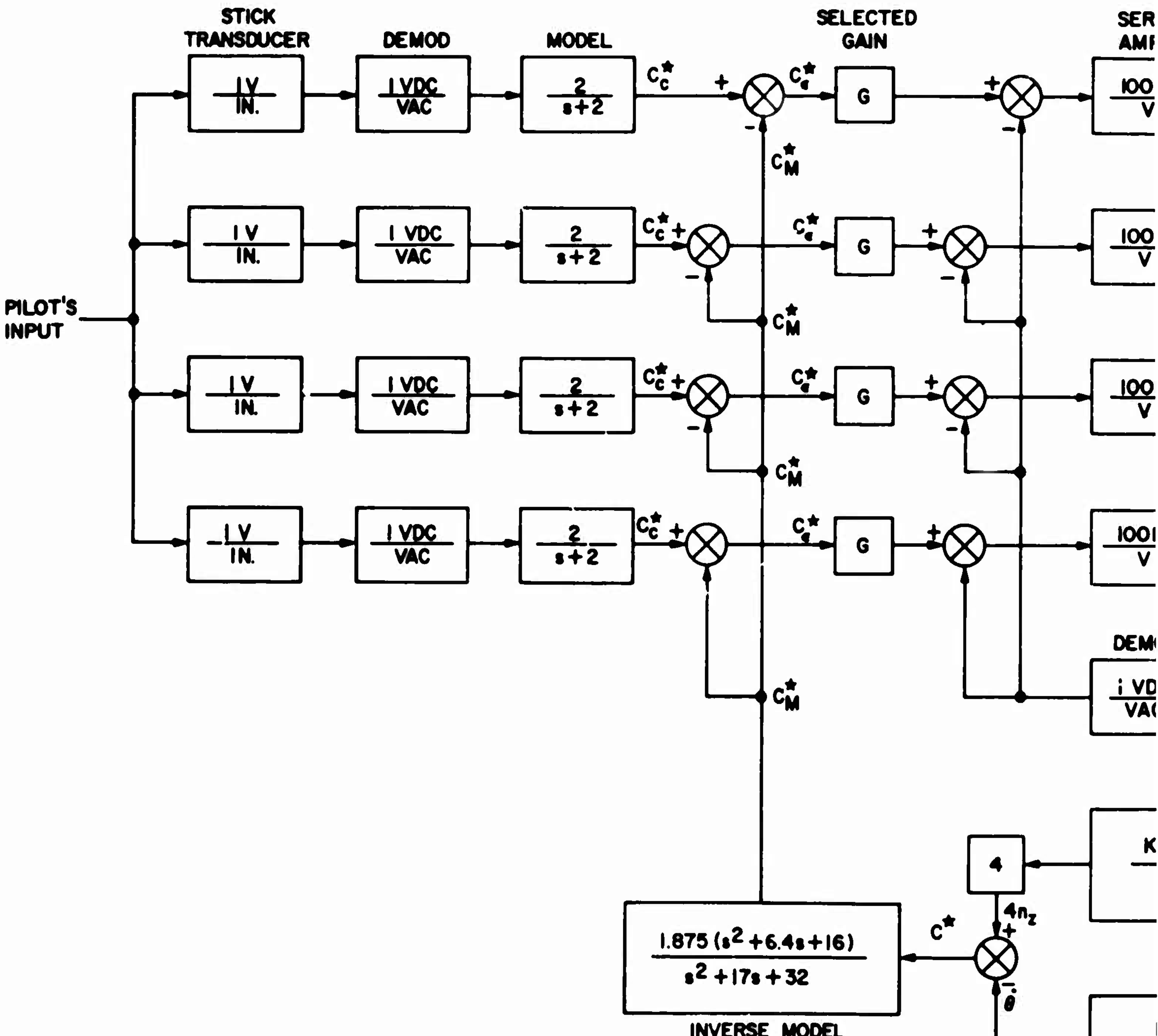


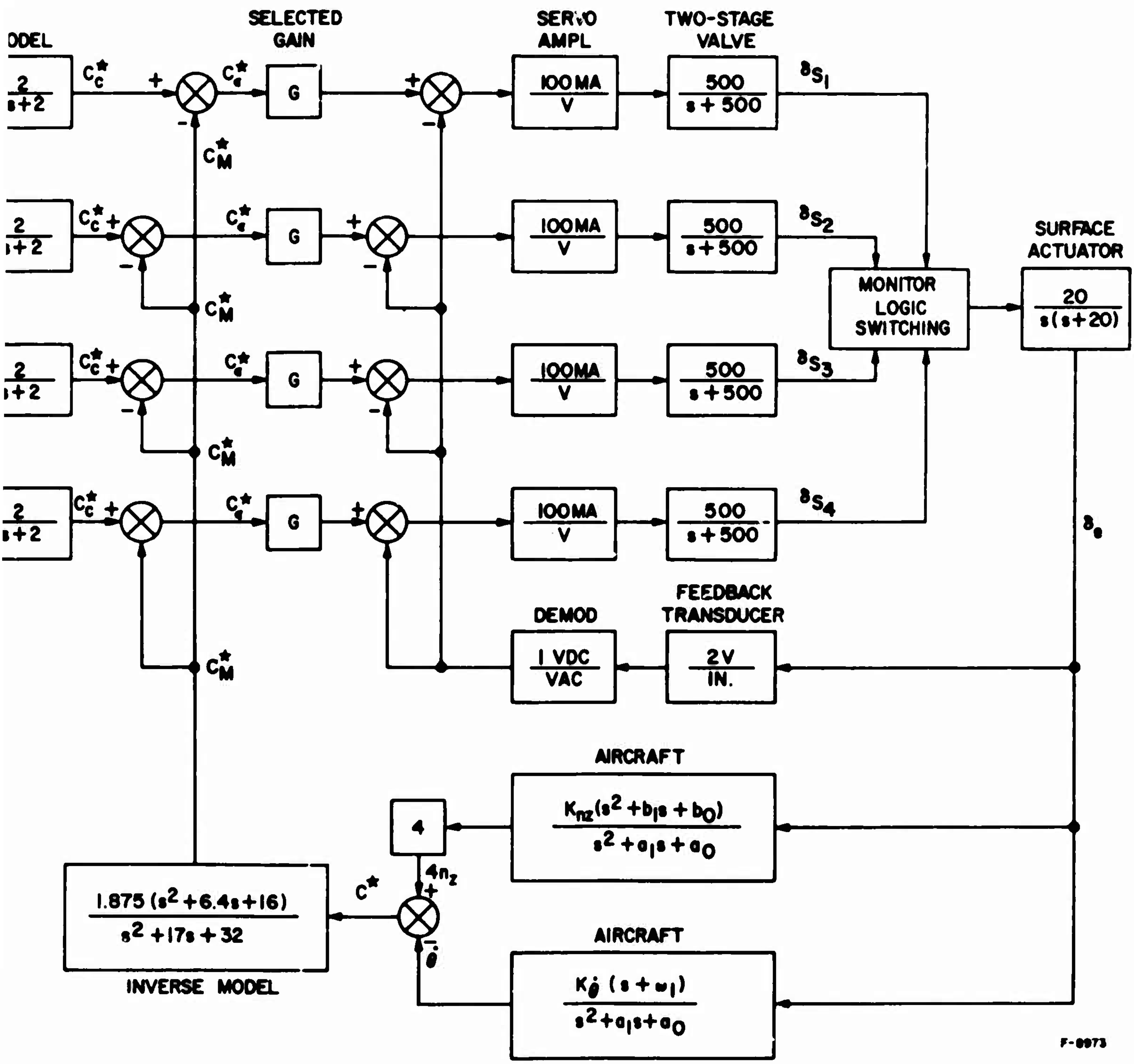


Figure 49
Block Diagram for Spool-Monitored Concept

12

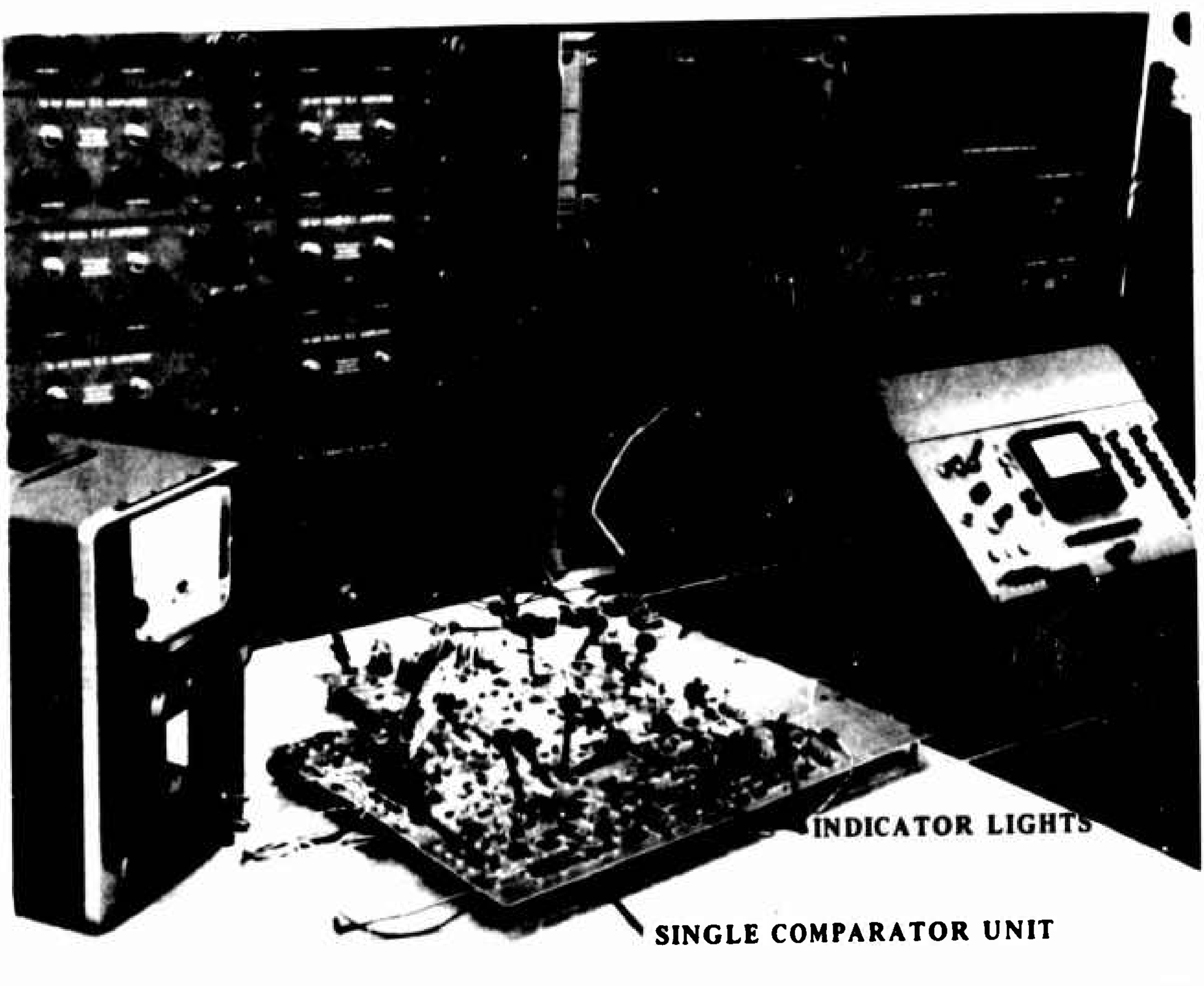


Figure 50
The Breadboard Simulation

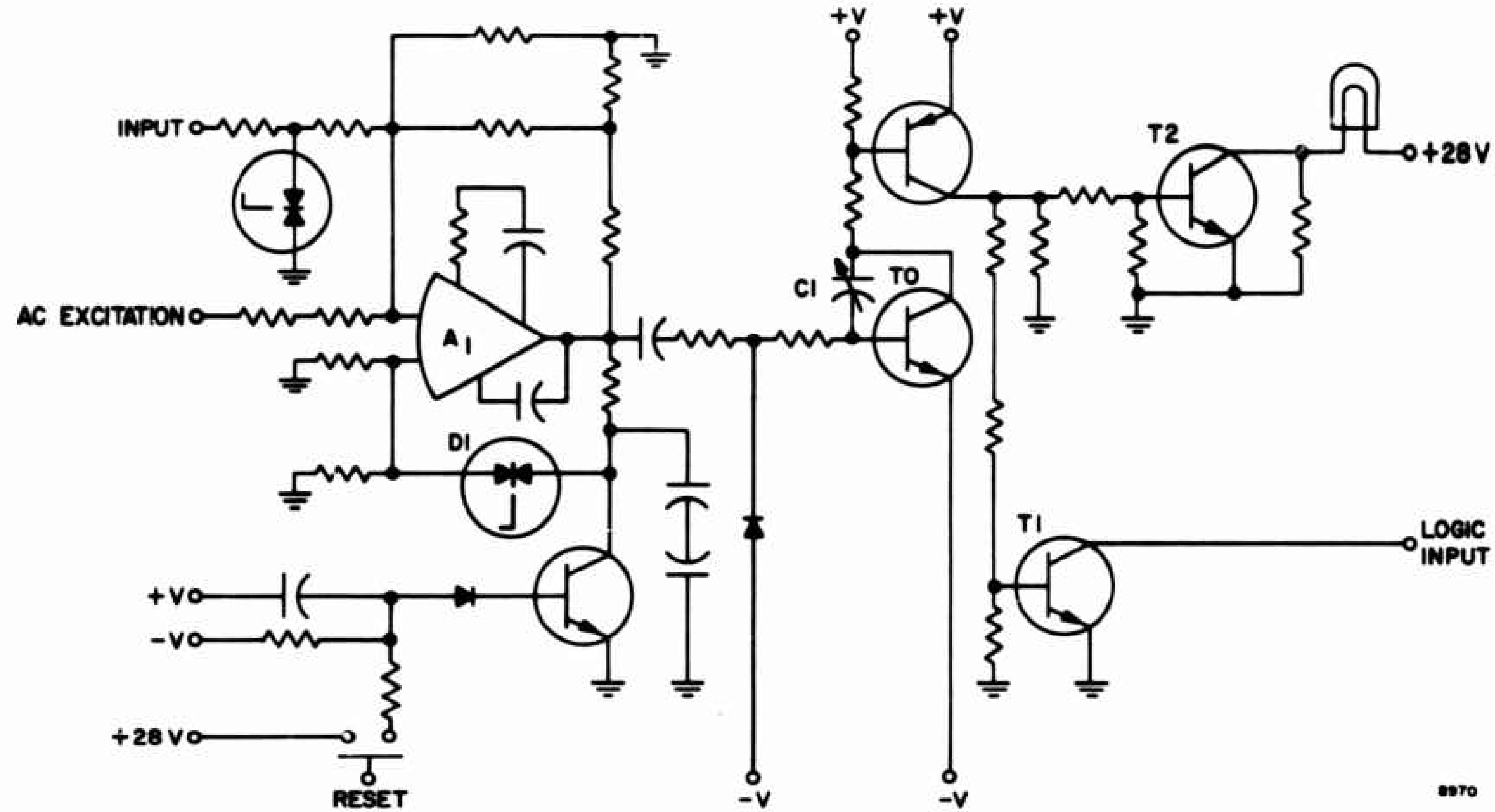


Figure 51
Fail-Safe Comparator

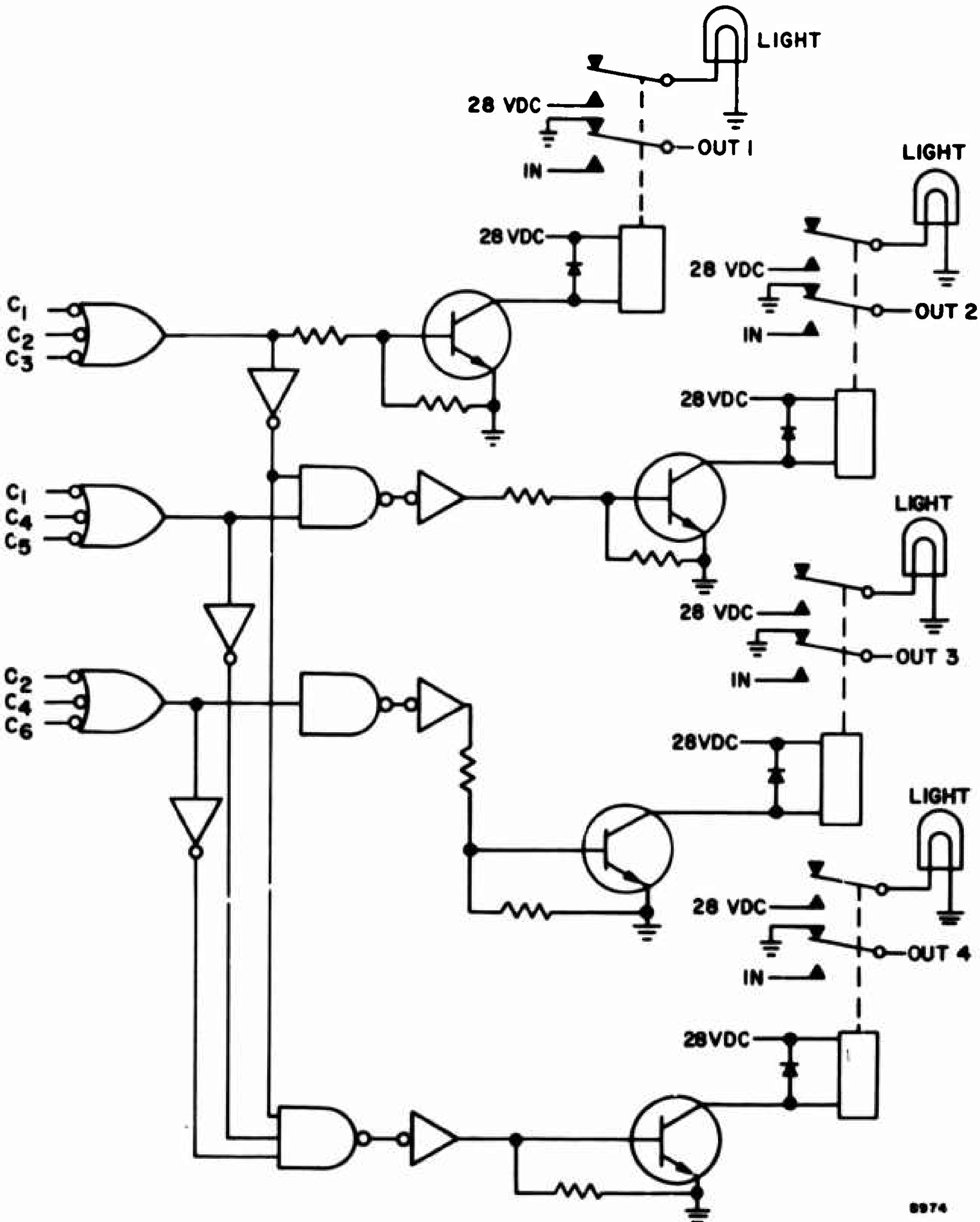


Figure 52
Logic and Switching Diagram

The relays employed in the breadboard model are used only to provide switching of the high voltages present in analog computation and would be replaced by solid-state transistor switches in flightworthy hardware.

2. SECONDARY ACTUATOR APPROACH

The secondary actuator (model 3) approach to a two-fail-operational actuator design is a descendant of the control systems employing series SAS servos driving a main surface actuator. For application to fly-by-wire, the system must have at least three independent channels (each capable of driving the surface actuator) and an electronic model channel. Monitoring, logic, and channel selection of the standby channels and the operating channel provide two-fail-operational capability with positive center and lock after a third failure. Figure 53 shows an analog diagram for the intermediate actuator concept. Potentiometer settings for the two flight conditions investigated are shown in table VI. The equations describing this simulation are as follows:

$$C^*_c = \left(\frac{2}{s+2}\right) \delta_s \tag{19}$$

$$I = (100G)\, C^*_\epsilon - 1000\, \delta_i \tag{20}$$

$$q = (0.0125)\, I \tag{21}$$

$$\delta_i = \left(\frac{4.55}{s}\right) q \tag{22}$$

$$\delta_e = \left(\frac{400}{s^2 + 20\, s + 400}\right) \delta_i \tag{23}$$

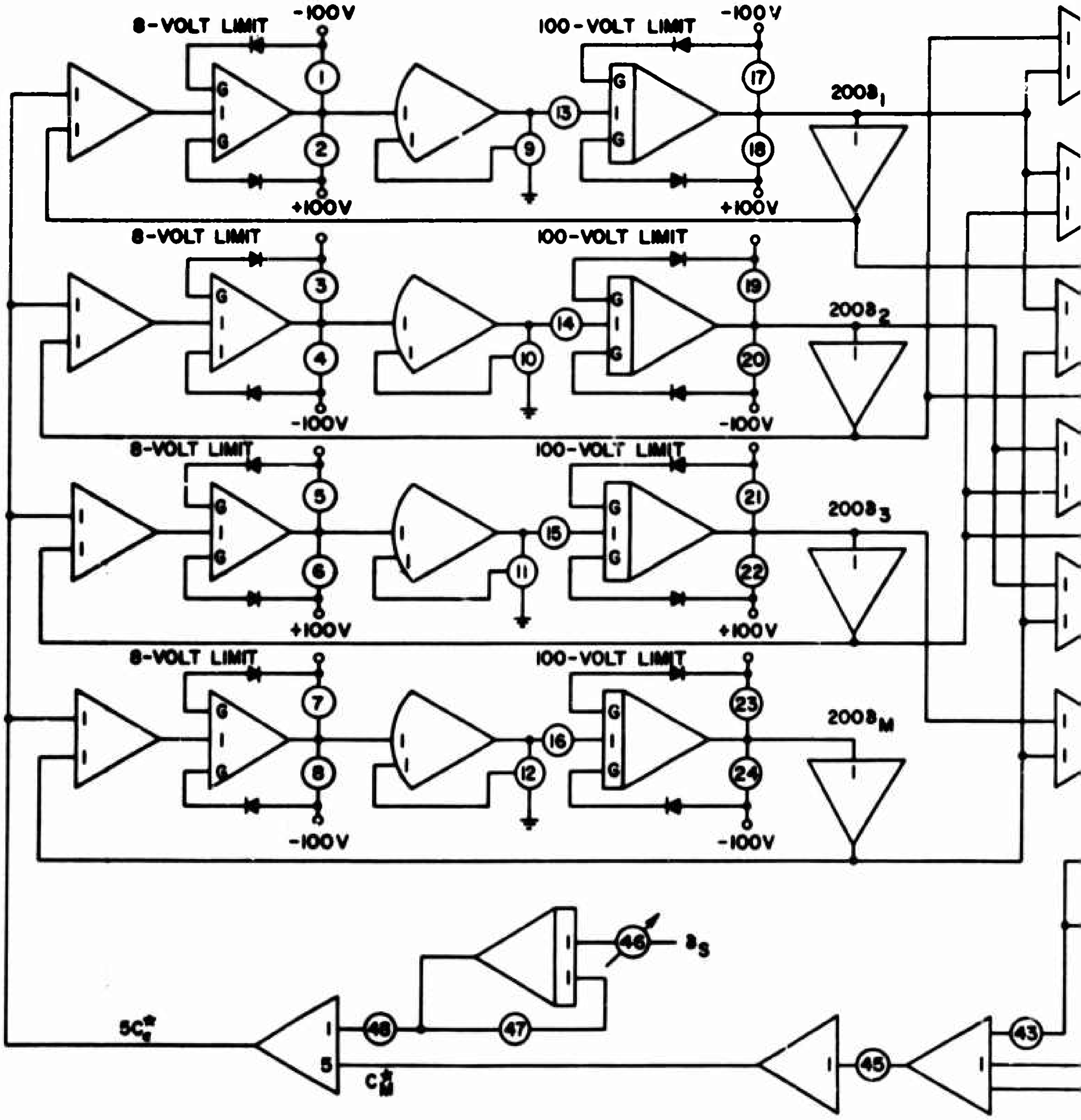
$$\dot{\theta} = \left(\frac{K_{\dot{\theta}}\, (s + \omega_1)}{s^2 + a_1\, s + a_o}\right) \delta_e \tag{24}$$

$$n_Z = \left(\frac{K_{n_Z}\, (s^2 + b_1\, s + b_o)}{s^2 + a_1\, s + a_o}\right) \delta_e \tag{25}$$

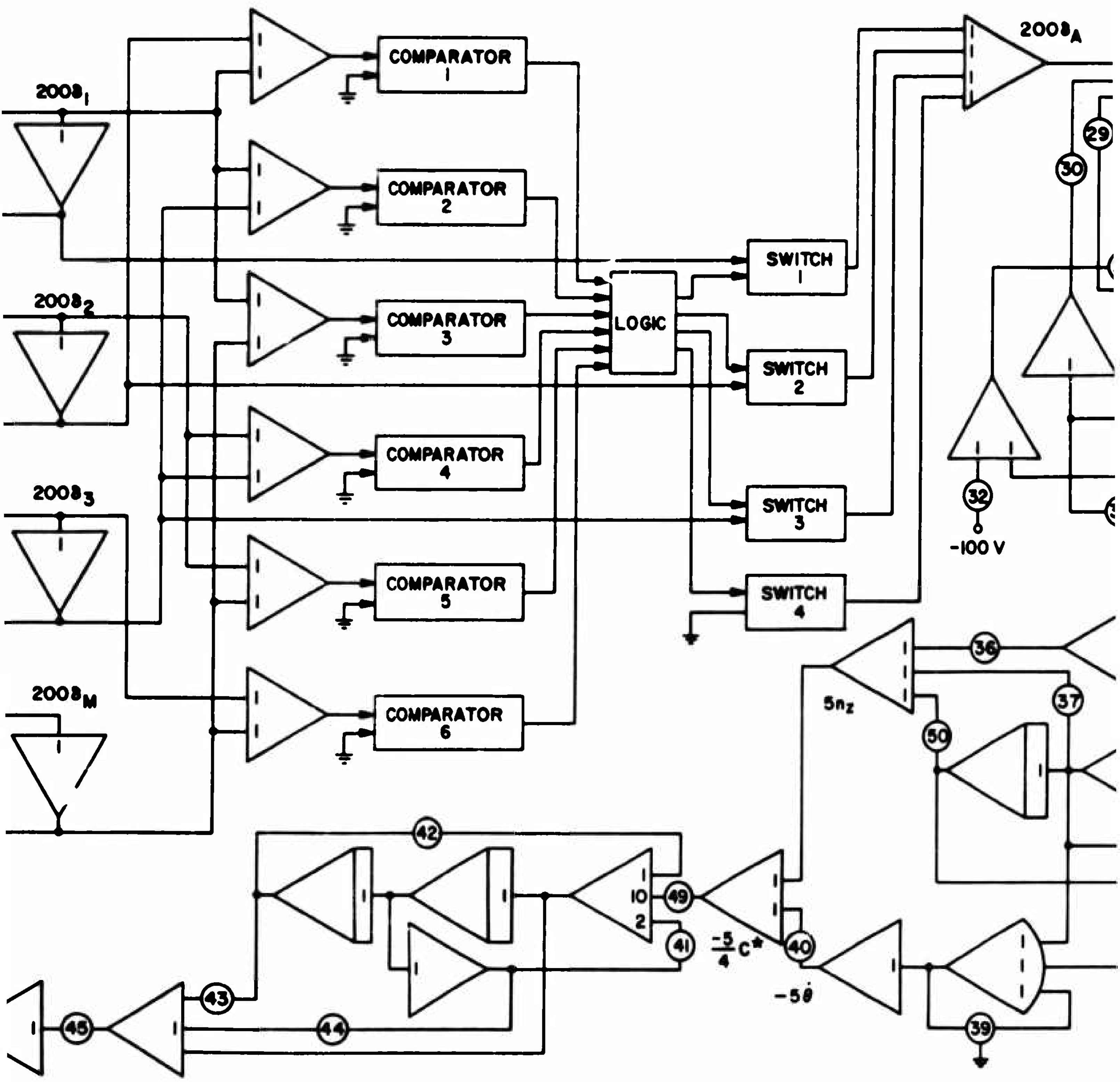
$$C^* = -\dot{\theta} + 4n_Z \tag{26}$$

$$C^*_M = \left(\frac{1.875\, (s^2 + 6.4\, s + 16)}{s^2 + 17s + 30}\right) C^* \tag{27}$$

$$C^*_\epsilon = C^*_c - C^*_M \tag{28}$$



8-VOLT LIMIT
-100V
100-VOLT LIMIT
-100V
+100V
+100V
200θ_1
200θ_2
200θ_3
200θ_M
-100V
+100V
a_s
$5C_s^*$
C_M^*


200δ1
200δ2
200δ3
200δM
COMPARATOR 1
COMPARATOR 2
COMPARATOR 3
COMPARATOR 4
COMPARATOR 5
COMPARATOR 6
LOGIC
SWITCH 1
SWITCH 2
SWITCH 3
SWITCH 4
200δA
29
30
32
-100 V
36
37
50
5nz
42
49
41
40
10
2
-5/4 c*
-5θ̇
39
43
44
45

Fi

Secon

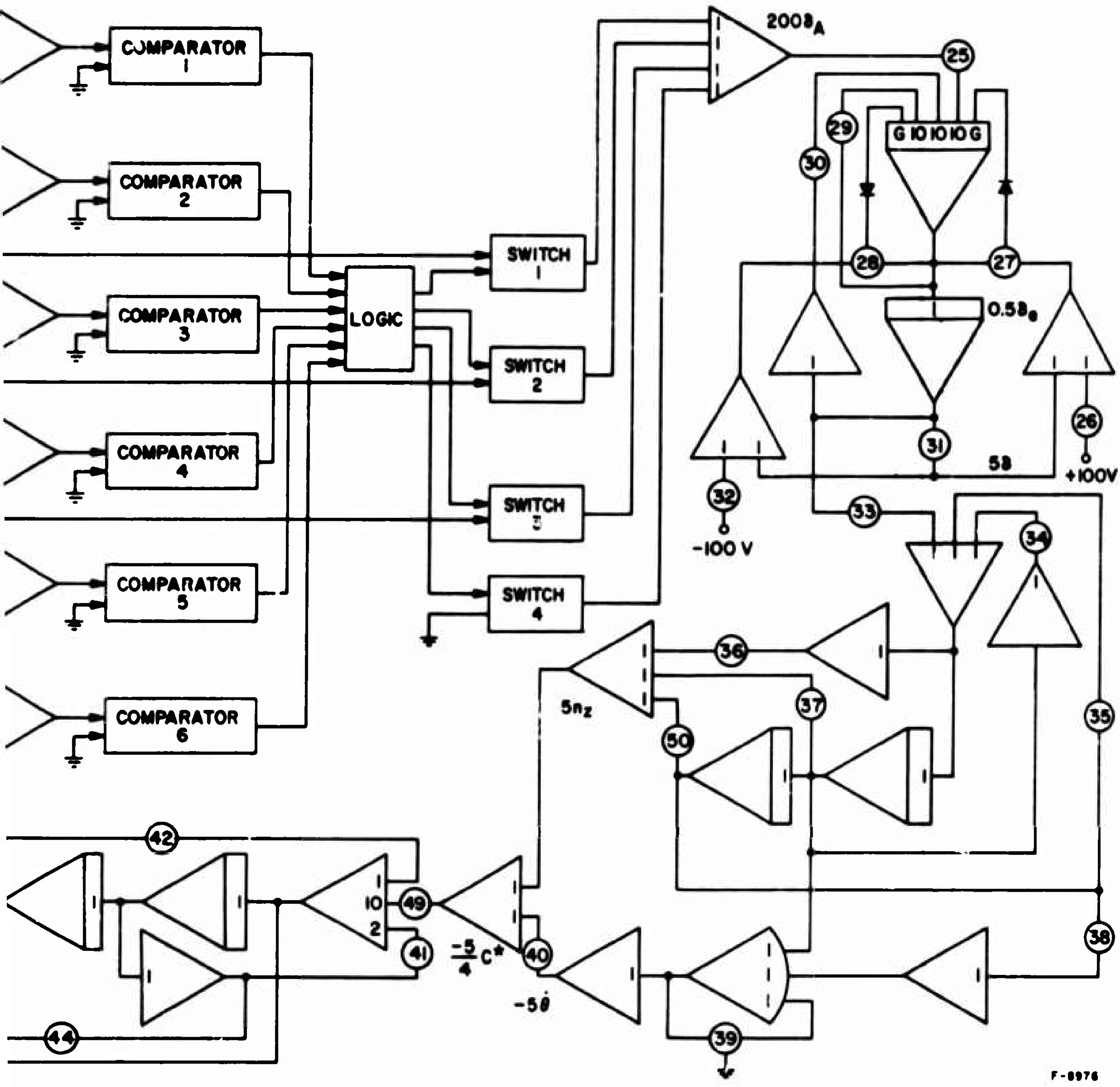


Figure 53
Secondary Actuator Concept
Analog Diagram

TABLE VI

SECONDARY ACTUATOR SIMULATION POTENTIOMETER SETTINGS

Pot Number	Setting	Function	Pot Number	Setting	Function
1	0.068		31	0.4	
2	0.068		32	0.2	
3	0.068	Limits on	33A	0.282 HQ	
4	0.068	Servo	33B	0.127 LQ	
5	0.068	Amplifier	34A	0.4248 HQ	
6	0.068	Current	34B	0.0753 LQ	Airframe
7	0.068		35A	0.282 HQ	Dynamics
8	0.068		35B	0.0048 LQ	
9	0.08	Servo-	36A	0.61 HQ	HQ = High
10	0.08	Valve/	36B	0.047 LQ	Dynamic
11	0.08	Secondary	37A	0.1616 HQ	Pressure
12	0.08	Actuator	37B	0.002 LQ	
13	0.455		38A	0.1333 HQ	LQ = Low
14	0.455		38B	0.0305 LQ	Dynamic
15	0.455		39A	0.0725 HQ	Pressure
16	0.455		39B	1.0 LQ	
17	0.5		40A	0.25 HQ	
18	0.5		40B	0.25 LQ	
19	0.5	Secondary	41	0.85	
20	0.5	Actuator	42	0.30	Inverse
21	0.5	Position	43	0.16	Model Plus
22	0.5	Limit	44	0.64	Shaping
23	0.5		45	0.8	
24	0.5		46	Variable	
25	0.4		47	0.2	
26	0.2	Elevator	48	0.2	
27	0.5	Plus	49	0.1875	
28	0.5	Limits	50A	0.7679 HQ	
29	0.2	on Rate	50B	0.0003 LQ	
30	0.4				

The following limits were imposed on the simulated actuator: servo amplifier (40 ma limit), on the secondary actuators (0.5 inch), and on the surface actuator rate (40 degrees/second).

The following assumptions were made in development of the simulation of the secondary actuator mechanization.

a. Triplex hydraulic and quadruplex electrical supplies.

b. The artificial feel sensors (rate gyro, accelerometer, and shaping network) are triplex with monitors capable of providing the following control channels and the model with identical output signals.

(Since failures of the sensors were not being investigated, only a single unit was used in the simulation.)

c. The simulation results are not significantly affected by connecting the model pistons to operate as secondary actuators after a failure rather then switching the servovalve from the model piston to the standby piston on the secondary actuator. Figure 54 shows the actual and the simulated switching. Typical signals to the surface actuator are given for each case to demonstrate why high response characteristics for the secondary actuator are desirable.

d. The scheduled gain G is constant for the flight condition under investigation but is varied for different flight conditions to maximizing loop gain.

e. The demodulator, modulator, amplifier and torquer are linear in the frequency range of interest.

f. Linear two-degree-of-freedom airframe dynamics for a typical high performance aircraft are adequate for determining response transients.

The simulation is time scaled by a factor of 10 to allow inclusion of the higher order actuator dynamics and to allow longer effective switching times for the hardware (comparators) at hand.

The failures investigated in the simulation are divided into two groups: passive (those which produce no output signal), and active (those which produce hardover signals). Examples of passive failures are:

a. Loss of control stick transducer

b. Loss of hydraulic supply

c. Loss of electrical supply

d. Open electronics channel

Examples of active failures are:

a. Loss of feedback transducer

b. Stuck servovalve

c. Servo amplifier hardover

For simulation purposes, the passive failures are simulated as loss of stick transducer, and the active failures are simulated by the loss of feedback transducers.

Failure-induced aircraft transients are obtained for two flight conditions (high and low dynamic pressure) and three channel-transfer times (10, 25, and 50 milliseconds). Figures 55 through 58 present the time histories of selected variables at the low dynamic pressure flight conditions.

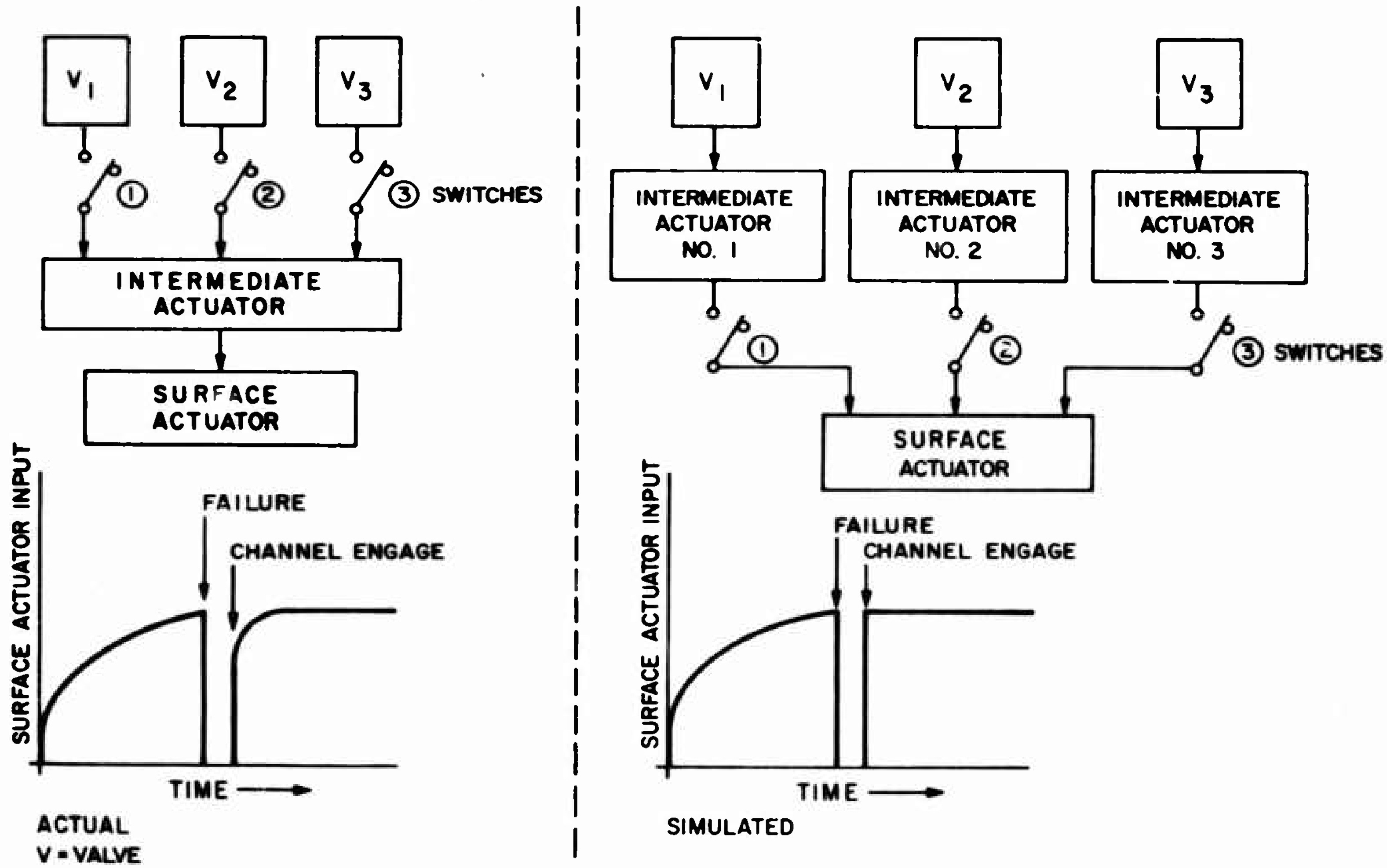


Figure 54
Comparison of Actual Switching and Simulated Switching
in the Secondary Actuator

Channels 1, 2, and 3 present the input command to the surface actuator from control channels 1, 2, and 3 respectively. Channel 4 shows the actual elevator position. Airframe normal acceleration and pitch attitude rate are shown on channels 5 and 6. Figures 55, 56, and 57 show the influence of channel selection time on aircraft transient for passive failures. It is obvious when comparing the resulting aircraft transient, that channel selection time is not critical for low dynamic pressure flight conditions. Reference to aircraft transient on figures 58, 59, and 60, presenting the influence of channel selection time on active failures, again shows that channel selection is not critical at low dynamic pressures. The data on figures 55 through 58 also demonstrate the benefit of the C* feedback technique. A comparison of the step response at t = 0, where C* is employed, and the effective step response resulting from a third failure due to center and lock of the actuator, where C* is no longer available, shows that the dynamic response of the airframe is appreciably improved by the artificial feel package. This does not mean that a pilot could not fly the aircraft at low dynamic pressures without artificial feel, but that much less pilot effort would be required through the benefits derived from C* feedback.

Figures 61 through 66 present time histories of the same flight variables presented at low dynamic pressure, at the high dynamic pressure flight condition. This flight condition is the critical region with respect to airframe transients and channel selection time because aerodynamic forces and moments produced by small surface deflections are large. The aircraft is then highly responsive. Figures 61, 62, and 63 present the airframe transient resulting from passive failures with 10, 25, and 50 millisecond channel selection times.

The detect/engage aircraft transients are reduced from 0.2 g and 0.1 g in normal acceleration and from 1.5 degrees each second to 0.5 degree each second in pitch rate by reducing the channel selection time from 50 milliseconds to 10 milliseconds. The 10-millisecond channel selection time still results in normal acceleration transients greater than those considered acceptable (0.02 g) for advanced commercial aircraft. Figures 63, 64, and 65 present airframe transients resulting from active failures. The transients in normal acceleration and pitch rate are directly comparable to those resulting from passive failures. This result is pessimistic since linear airframe dynamics are assumed, and acceleration limiting is not included. Again, as at the low dynamic pressure region, the effects of C* feedback can be observed by comparing the initial step response at t = 0 to that occurring as a result of the third failure producing actuator center and lock.

At the high dynamic pressure condition, the artificial feel produces a larger effect in changing the aircraft damping than it does in changing the aircraft speed of response. This result is expected since the effective control frequency of the aircraft is a function dynamic pressure. In the high dynamic pressure region, the control response meets the C* response criteria except for the damping ratio.

The results of the simulation of the secondary actuator approach to fly-by-wire implementation indicate that the actuator design meets all design criteria providing the failure-induced airframe transients are acceptable. This actuator configuration provides double failure operation with positive center and lock capability if a third failure occurs. Airframe transients

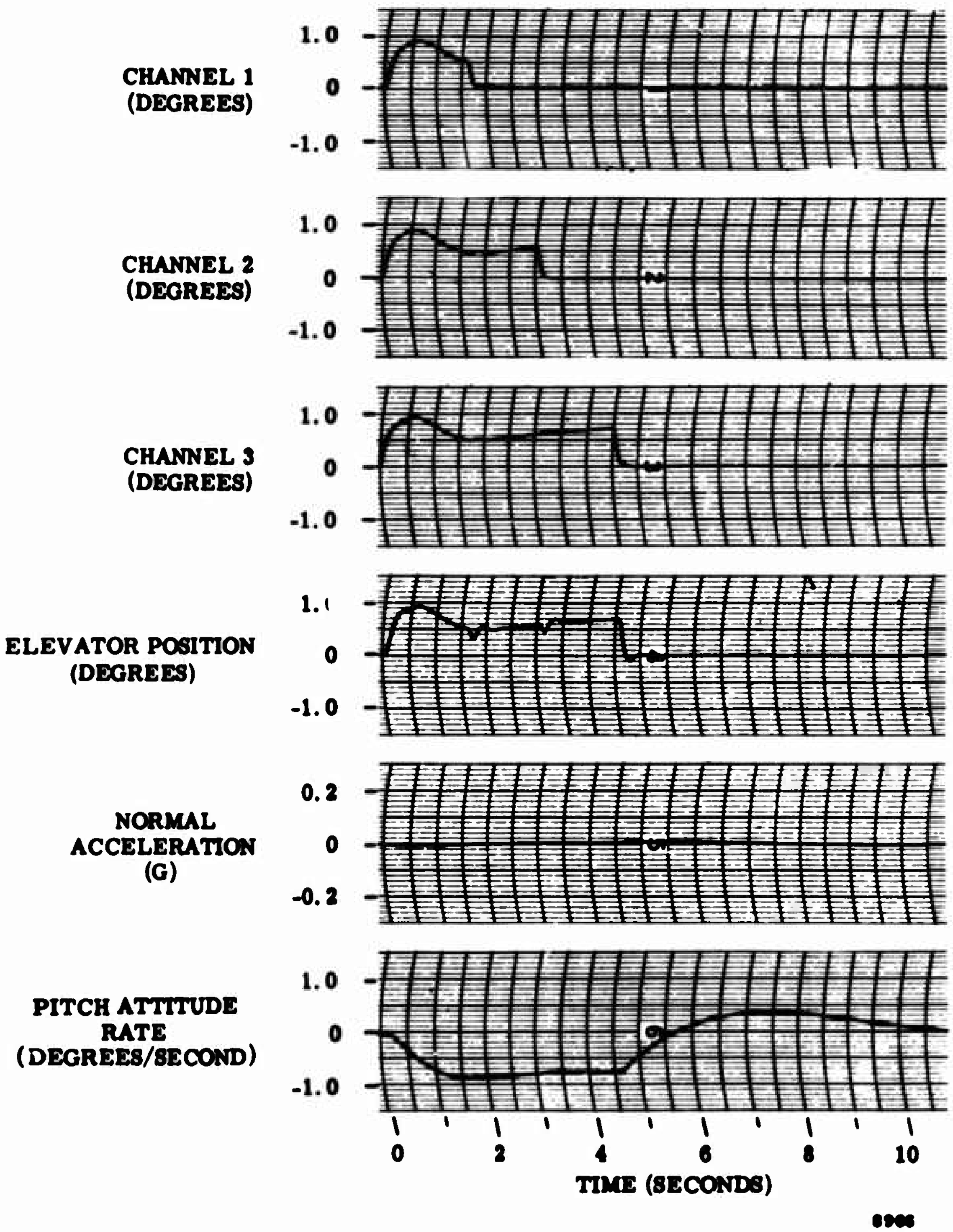


Figure 55
Secondary Actuator at Low Dynamic Pressure, 10 ms Switch/Open Input

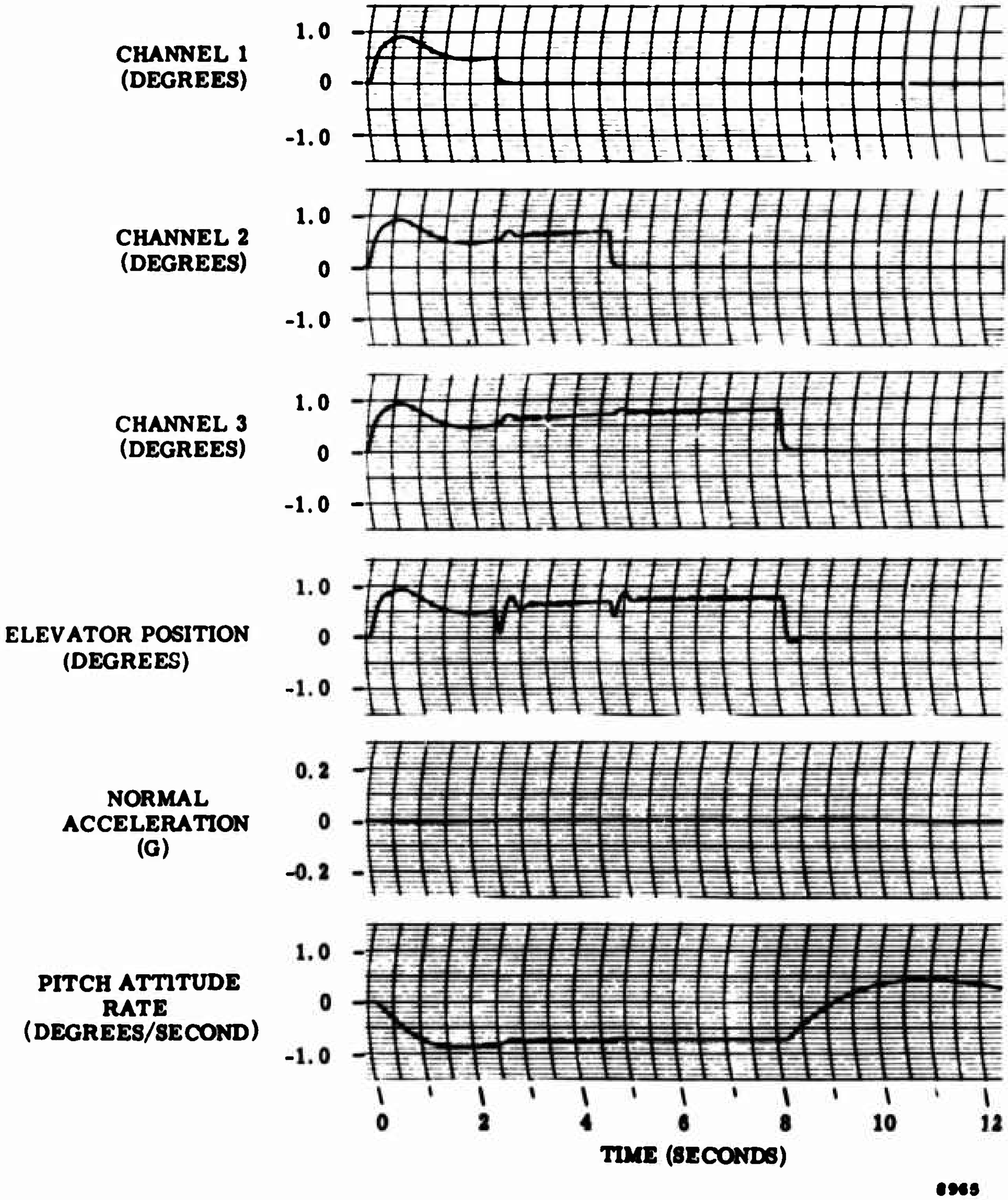


Figure 56
Secondary Actuator at Low Dynamic Pressure, 25 ms Switch/Open Input

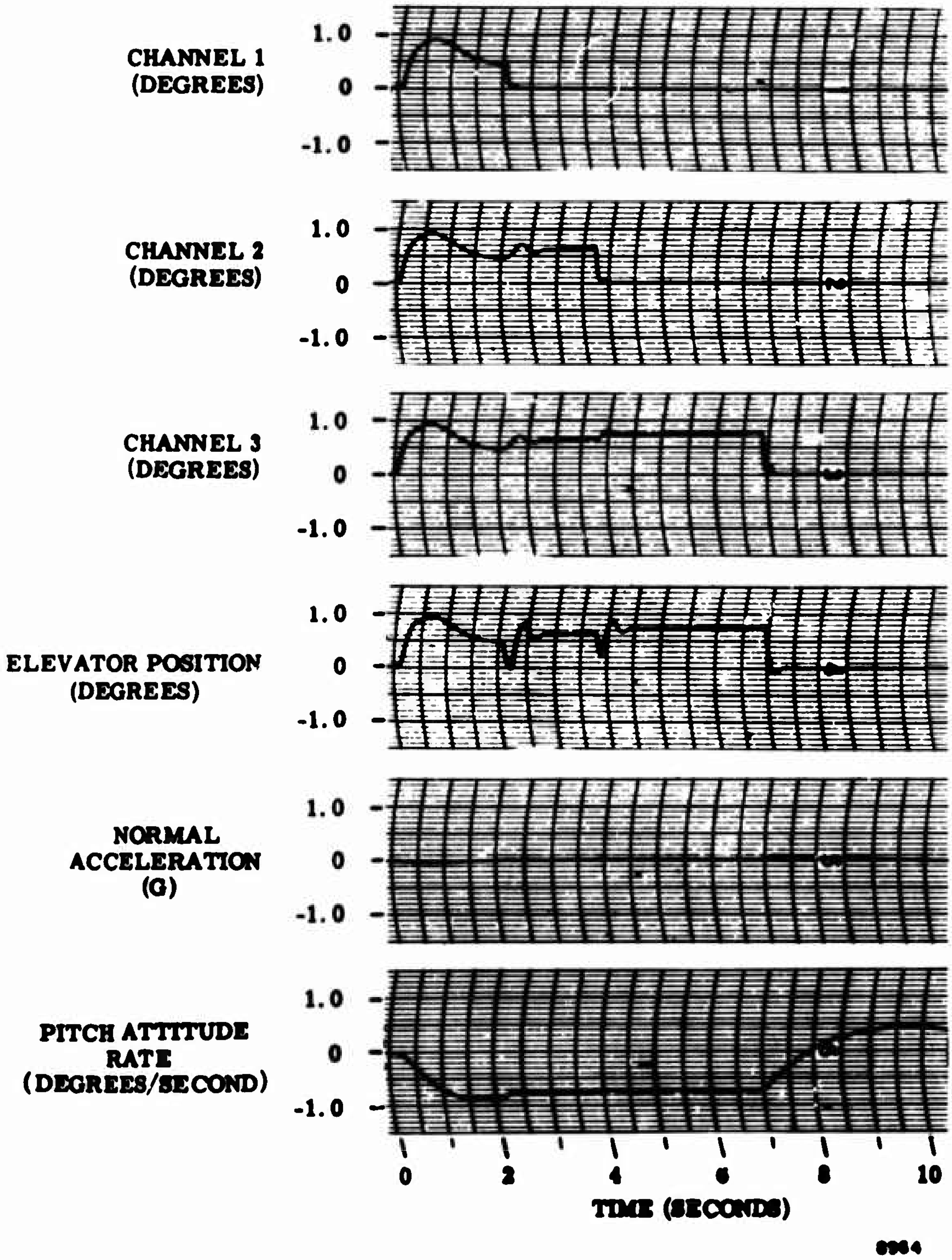


Figure 57
Secondary Actuator at Low Dynamic Pressure, 50 ms Switch/Open Input

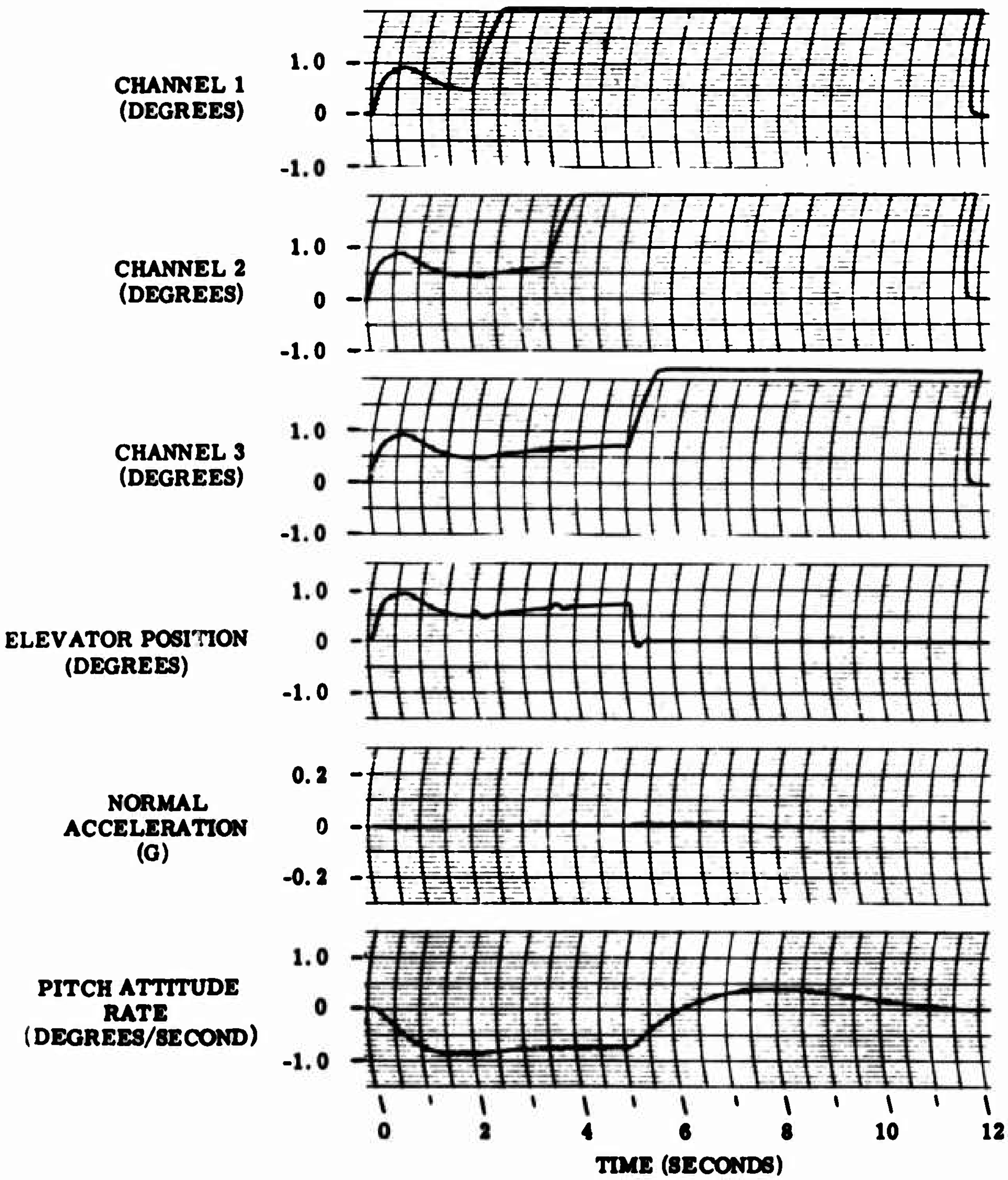


Figure 58
Secondary Actuator at Low Dynamic Pressure, 10 ms Switch/Open Feedback

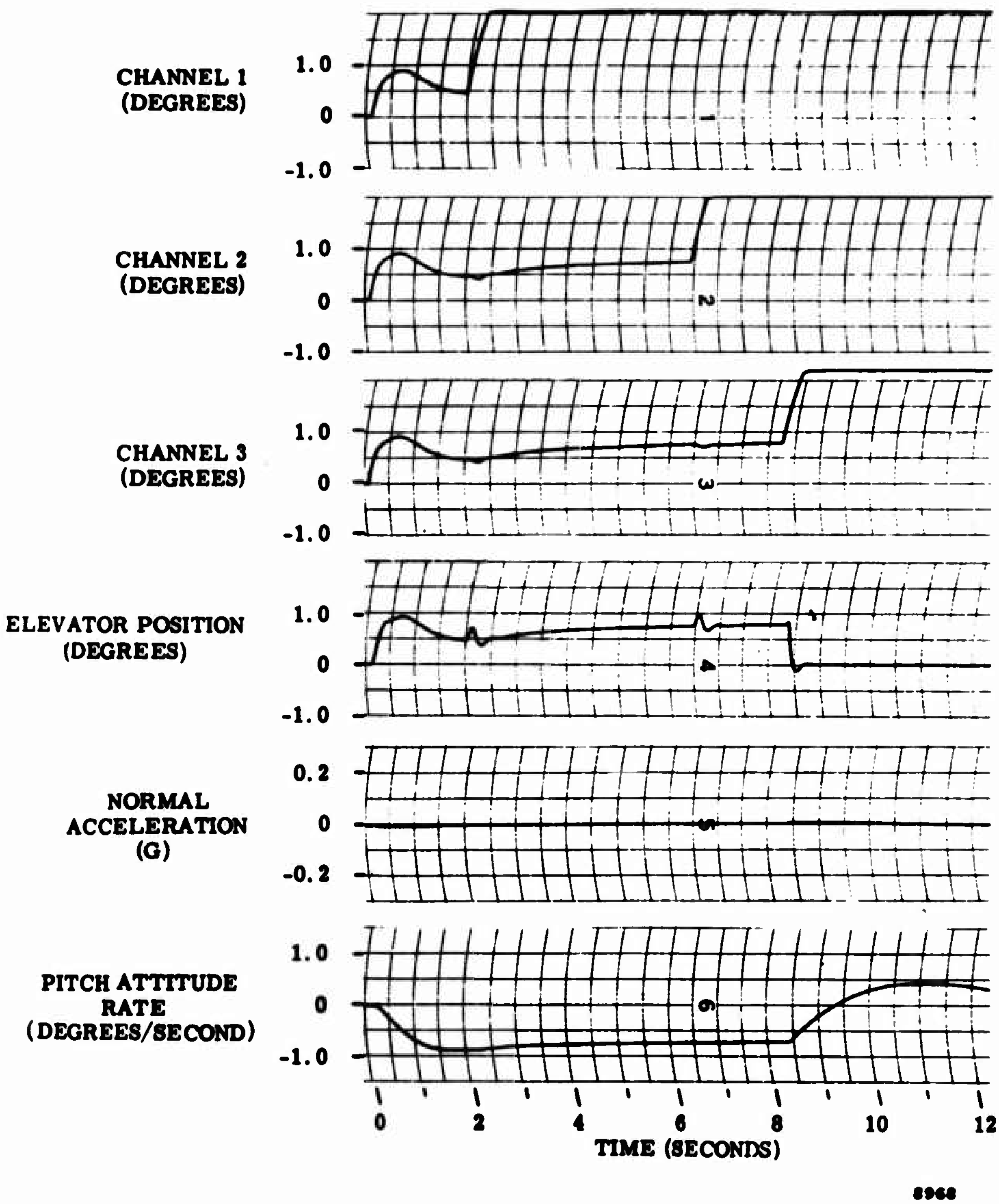


Figure 59
Secondary Actuator at Low Dynamic Pressure, 25 ms Switch/Open Feedback

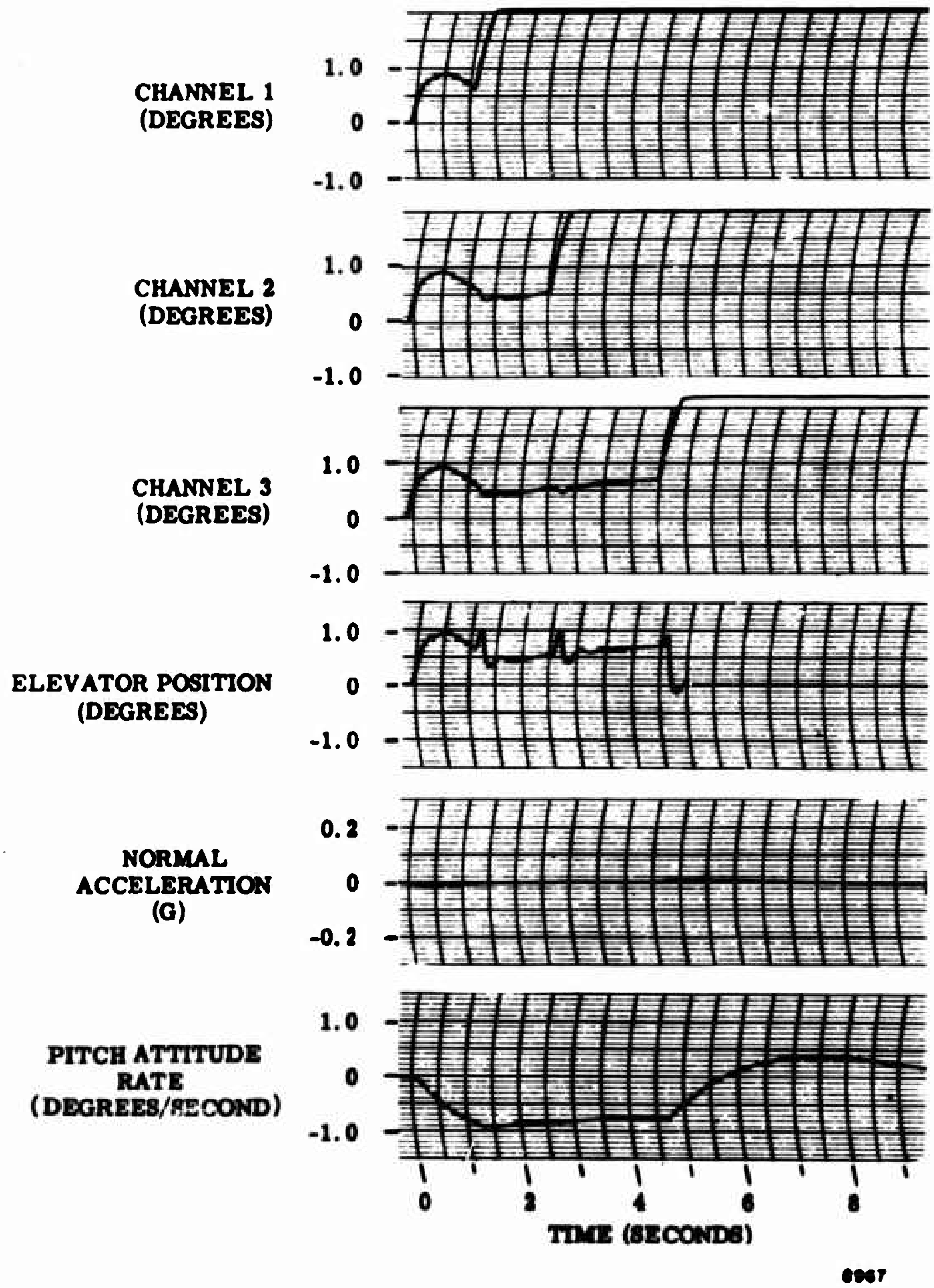


Figure 60
Secondary Actuator at Low Dynamic Pressure, 50 ms Switch/Open Feedback

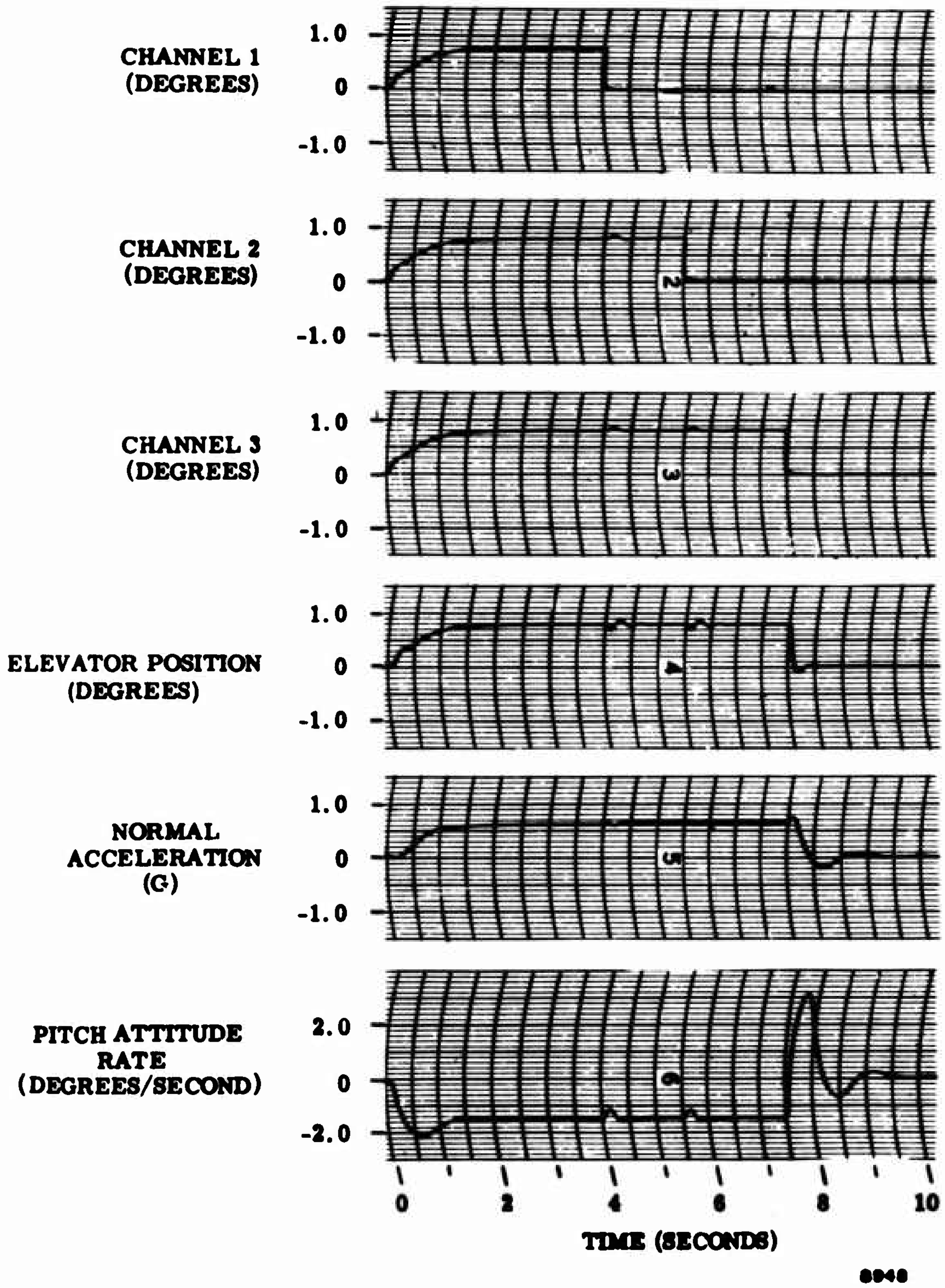


Figure 61
Secondary Actuator at High Dynamic Pressure, 10 ms Switch/Open Input

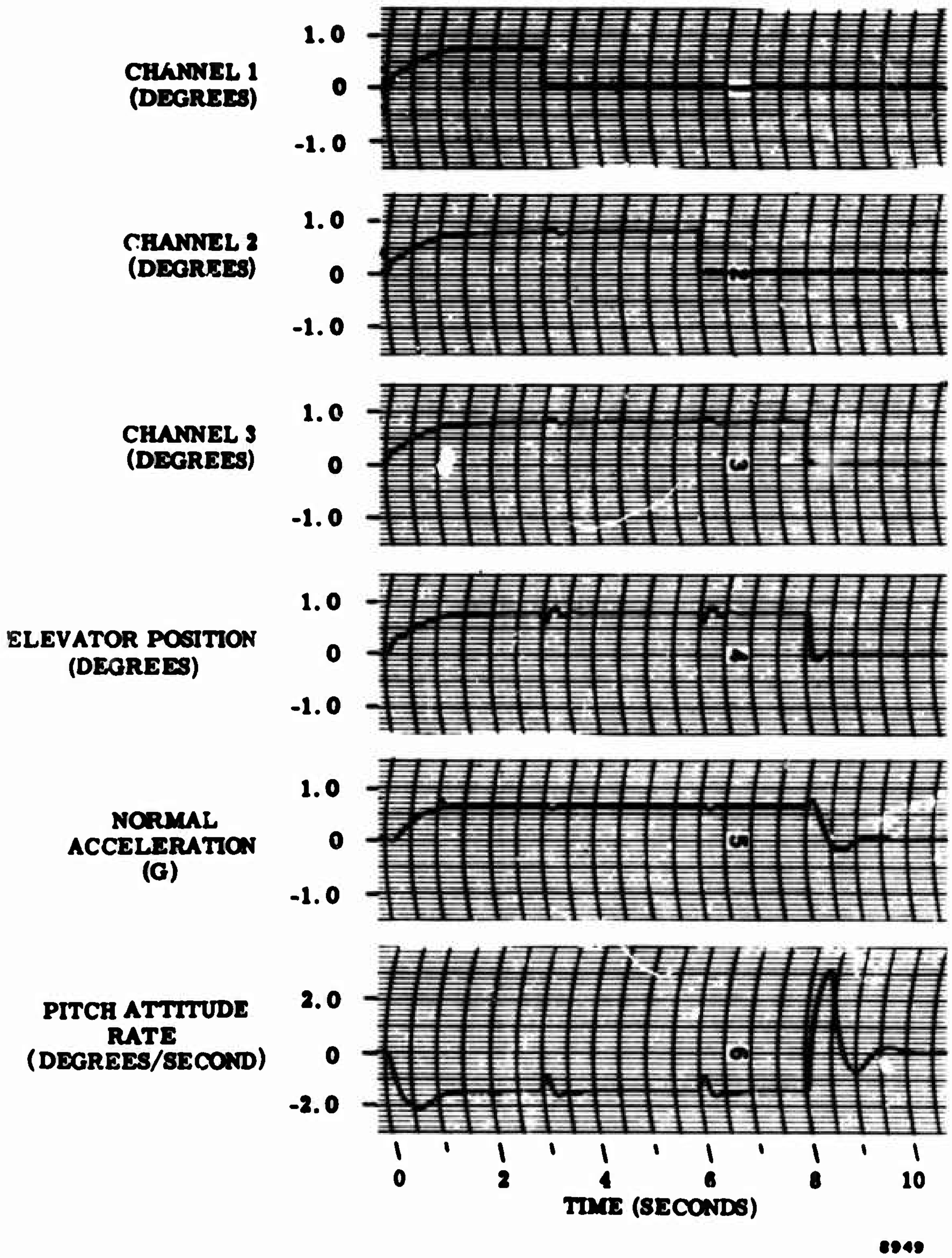


Figure 62
Secondary Actuator of High Dynamic Pressure, 25 ms Switch/Open Input

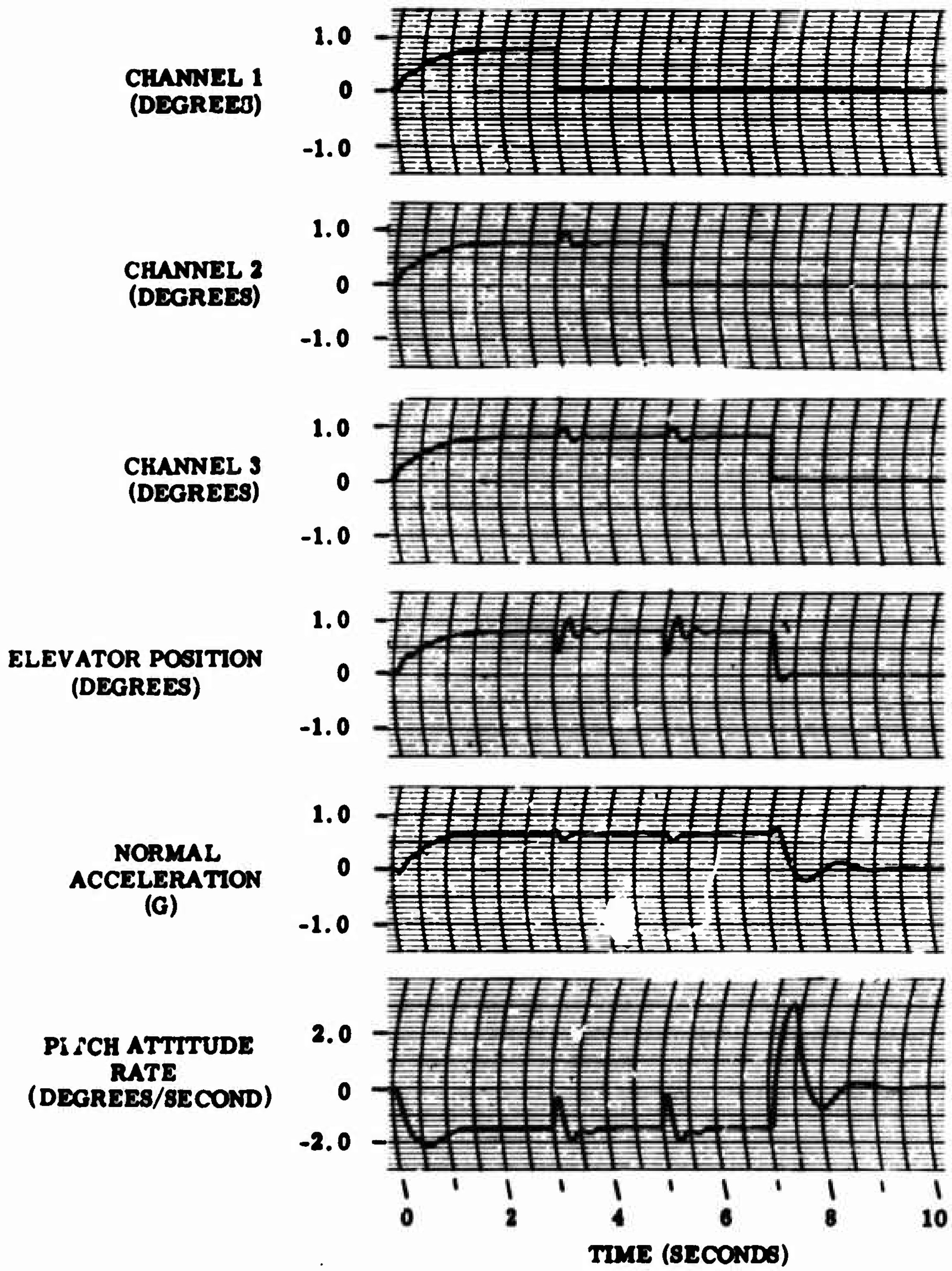


Figure 63
Secondary Actuator at High Dynamic Pressure, 50 ms Switch/Open Input

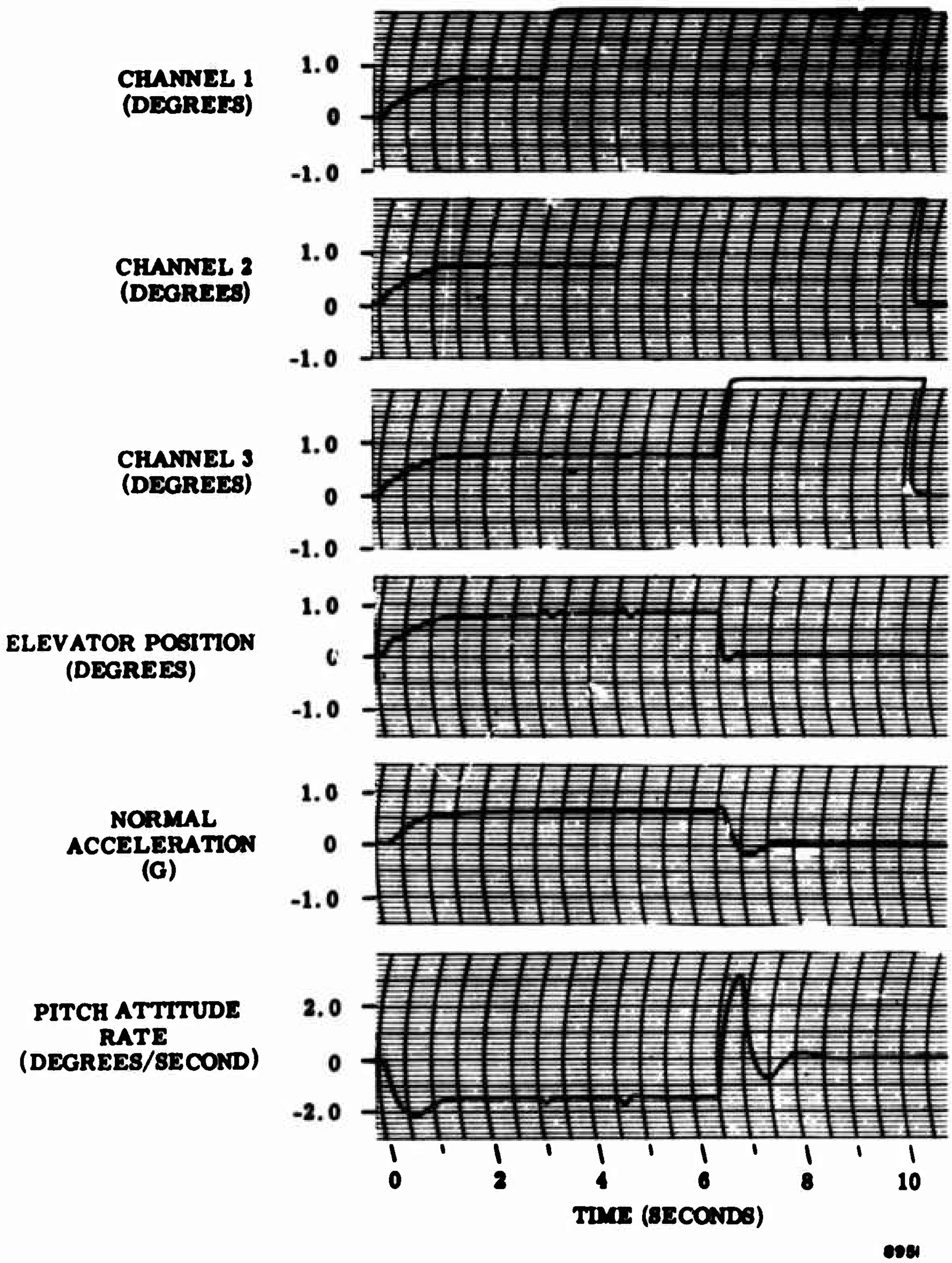


Figure 64
Secondary Actuator at High Dynamic Pressure, 10 ms Switch/Open Feedback

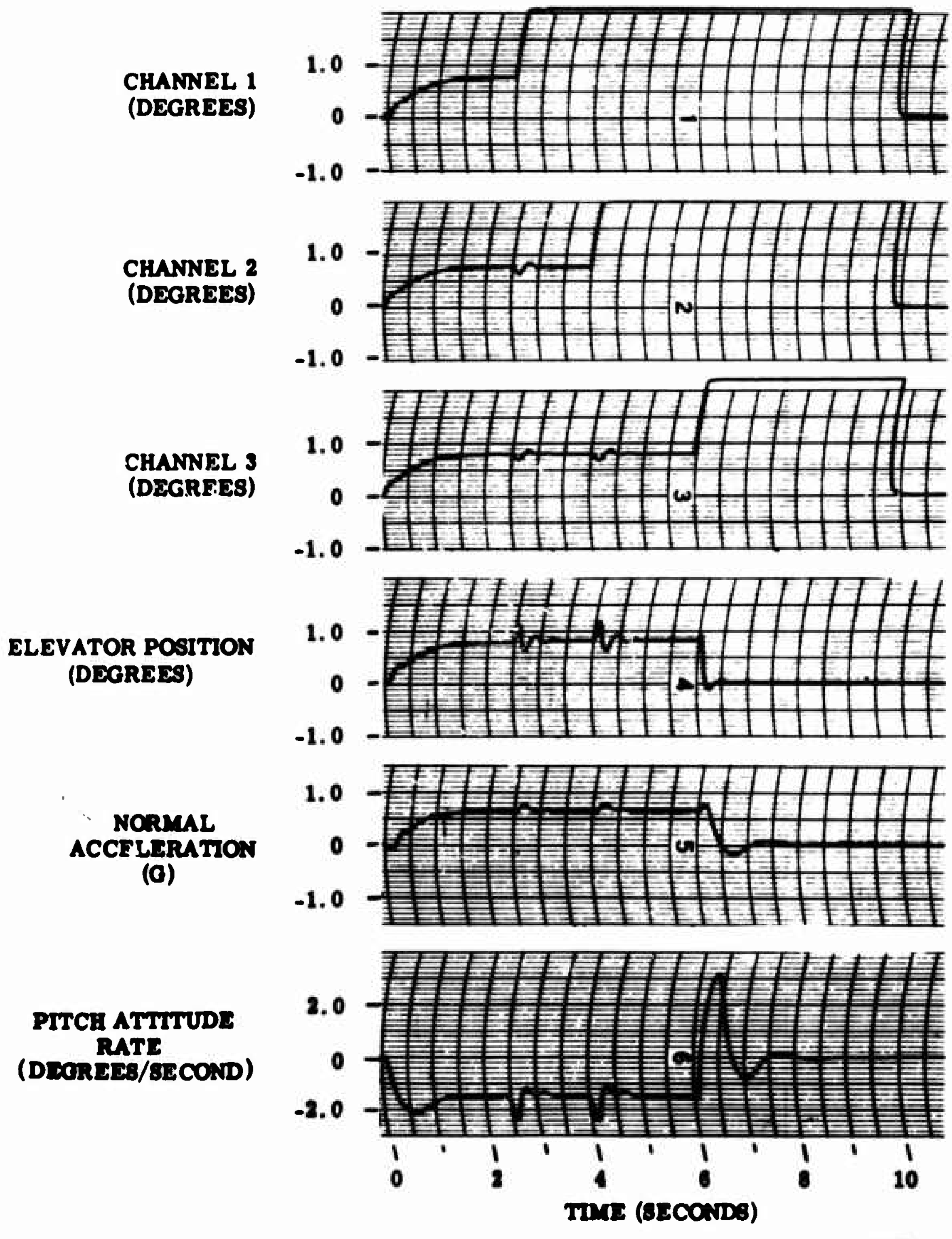


Figure 65
Secondary Actuator at High Dynamic Pressure, 25 ms Switch/Open Feedback

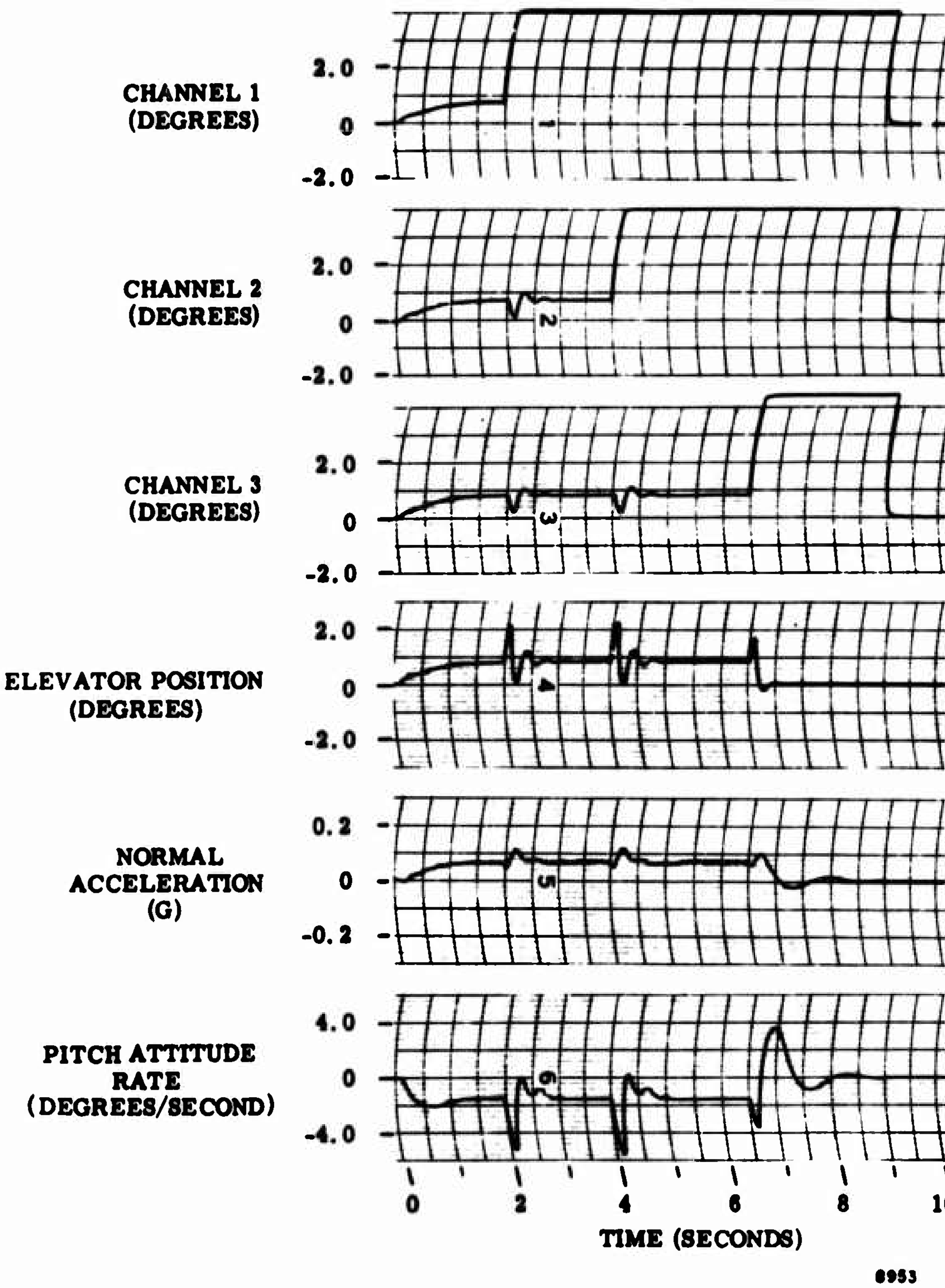


Figure 66
Secondary Actuator at High Dynamic Pressure, 50 ms Switch/Open Feedback

resulting from failure detection and channel engagement exceed those that the aircraft industry has set for advanced commercial aircraft such as the supersonic transport. However, the 0.02 g transient level for commercial aircraft primarily concerns passenger comfort and is set to prevent the passengers from detecting any control system failure through the airframe responses. Comfort of ride at this low g level and passenger failure awareness are not problems in military aircraft. Hence, a 0.1 to 0.2 g transient would probably be acceptable in military aircraft especially if this transient level only occurred at maximum dynamic pressure.

3. THE FAIL-PASSIVE ACTUATOR APPROACH

The fail-passive secondary actuator (model 4) approach to a two-fail-operational actuator design is a Sperry-pioneered design. The design objective is to optimize the application of the redundancy necessary to provide double failure operation with improvements in reliability and maintainability. Figures 67 and 68 show the electronic and hydraulic breadboard developed on company funds to analyze and evaluate fail-passive designs. The system utilizes position servos in which the electronic amplifiers, hydraulic servovalves, actuators, and position transducers can be made redundant to various degrees by changing a few simple electrical and/or mechanical connections. A simple aircraft simulator and servo-driven rate table were also fabricated to allow the demonstration of the fail-passive characteristic of a rate autopilot. With the addition of a normal accelerometer, this system would be identical to the fly-by-wire system using C* feedback.

The electronics test bed, shown in figure 67, is composed of three complete rate autopilot channels of electronics (a fourth channel, not shown, is also available). Each channel contains means for injecting faults. A simple aircraft, consisting of an analog computer and a servo-driven rate table, allows the demonstration of closed-loop operation of the autopilot.

The electrohydraulic test bed, shown in figure 68, is composed of three actuators on which jet-pipe valves are mounted. Provision is made for mounting two valves on each actuator to demonstrate the fail-passive characteristics of dual valve actuator systems. Each actuator is instrumented with differential pressure transducers for monitoring the pressure across the piston.

The actuators are shown in the force summation configuration. The loading fixture sums the actuator outputs through short, rigid links attached to a common output shaft. Thus, the actuators have a common position and their output forces are summed. The output shaft is implemented with synchro position feedback. Each actuator can be disconnected so that any combination of actuators can be operated at any one time. A fourth rigid link orthogonal to the output shaft is provided for attaching various spring and damper loads to the actuators.

The fail-passive concept for fly-by-wire application required three active and independent channels force summed at a secondary actuator. Monitoring is required to provide positive center and lock capability and for failure reporting. An analog diagram for the fail-passive actuator simulation is shown in figure 69. Potentiometer settings for the two flight

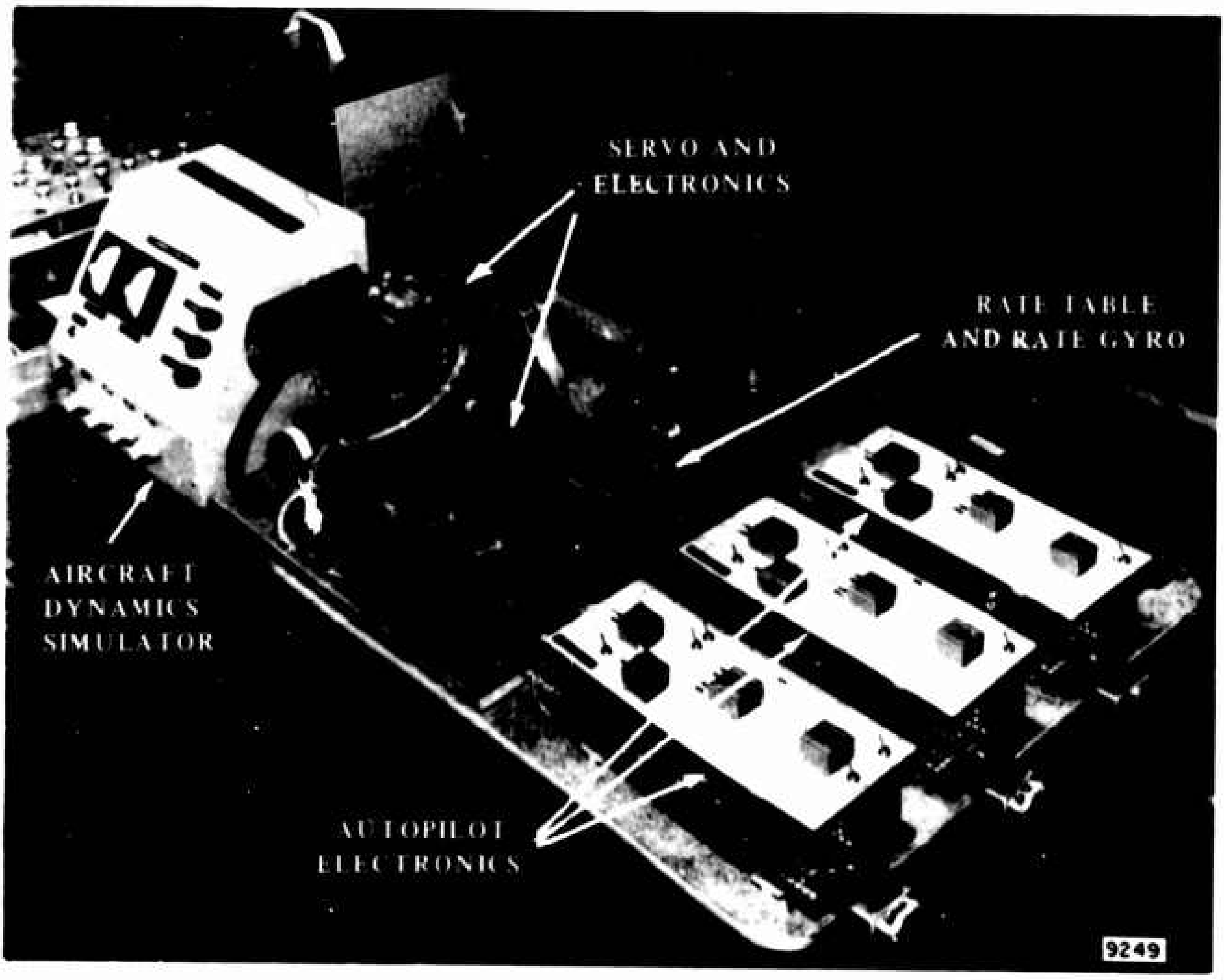


Figure 67
Electronics Test Bed

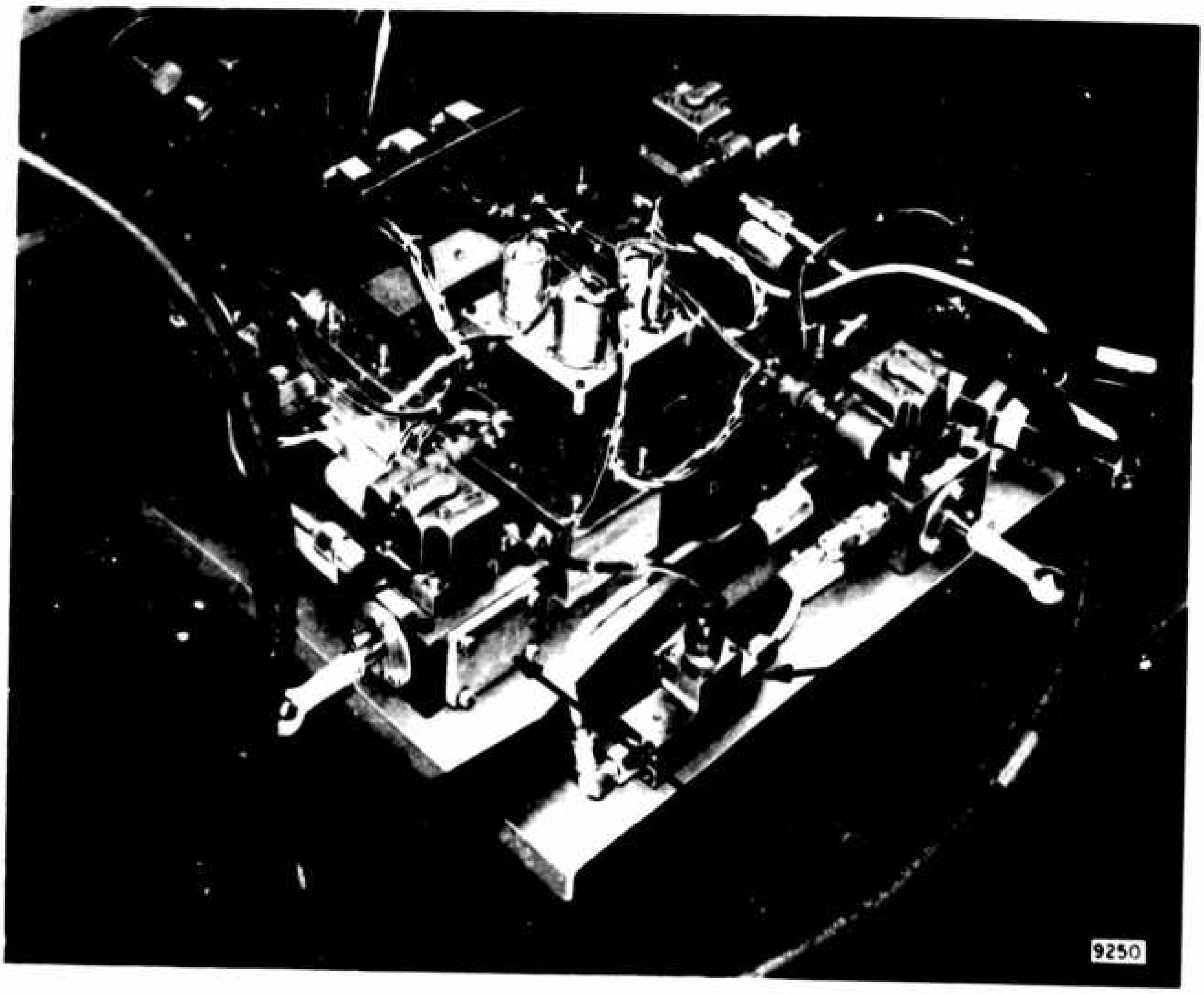


Figure 68
Electrohydraulic Test Bed

conditions investigated are shown in table VII. The equations describing the simulation are as follows:

$$C^*_c = \left(\frac{2}{S + 2}\right) \delta_s \tag{29}$$

$$I = (100G)\, C^*_\epsilon - 100\, \delta_A \tag{30}$$

$$F_i = 5\, I \tag{31}$$

$$\ddot{\delta}_A = \frac{1}{M} \sum F_i - \frac{P}{q} A^2 \dot{\delta}_A \tag{32}$$

$$\delta_e = \left(\frac{400}{s^2 + 20 + 400}\right) \delta_A \tag{33}$$

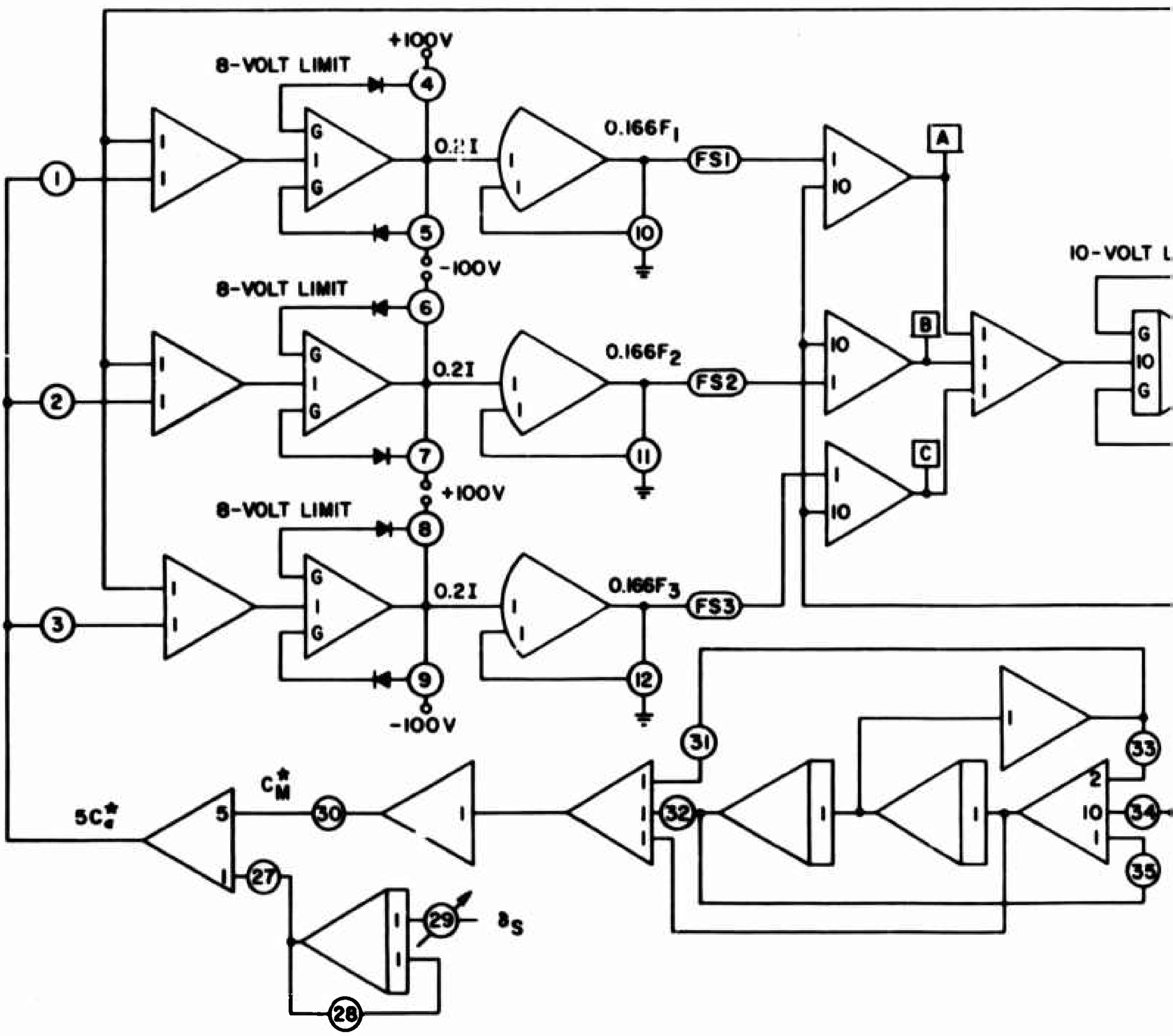
$$\dot{\theta} = \frac{K_{\dot{\theta}}\,(s + \omega_1)}{s^2 + a_1 s + a_o} \delta_e \tag{34}$$

$$n_Z = \frac{K_{n_z}\,(s^2 + b_1 s + b_o)}{s^2 + a_1 s + a_o} \delta_e \tag{35}$$

$$C^* = -\dot{\theta} + 4 n_z \tag{36}$$

$$C^*_M = \frac{1.875\,(s^2 + 6.4\, s + 16)}{s^2 + 17\, s + 30} C^* \tag{37}$$

$$C^*_\epsilon = C^*_c - C^*_M \tag{38}$$


8-VOLT LIMIT
+100V
0.2I
0.166F1
FS1
-100V
8-VOLT LIMIT
0.2I
0.166F2
FS2
+100V
8-VOLT LIMIT
0.2I
0.166F3
FS3
-100V
10-VOLT L
A
B
C
5C*
C*M
δS

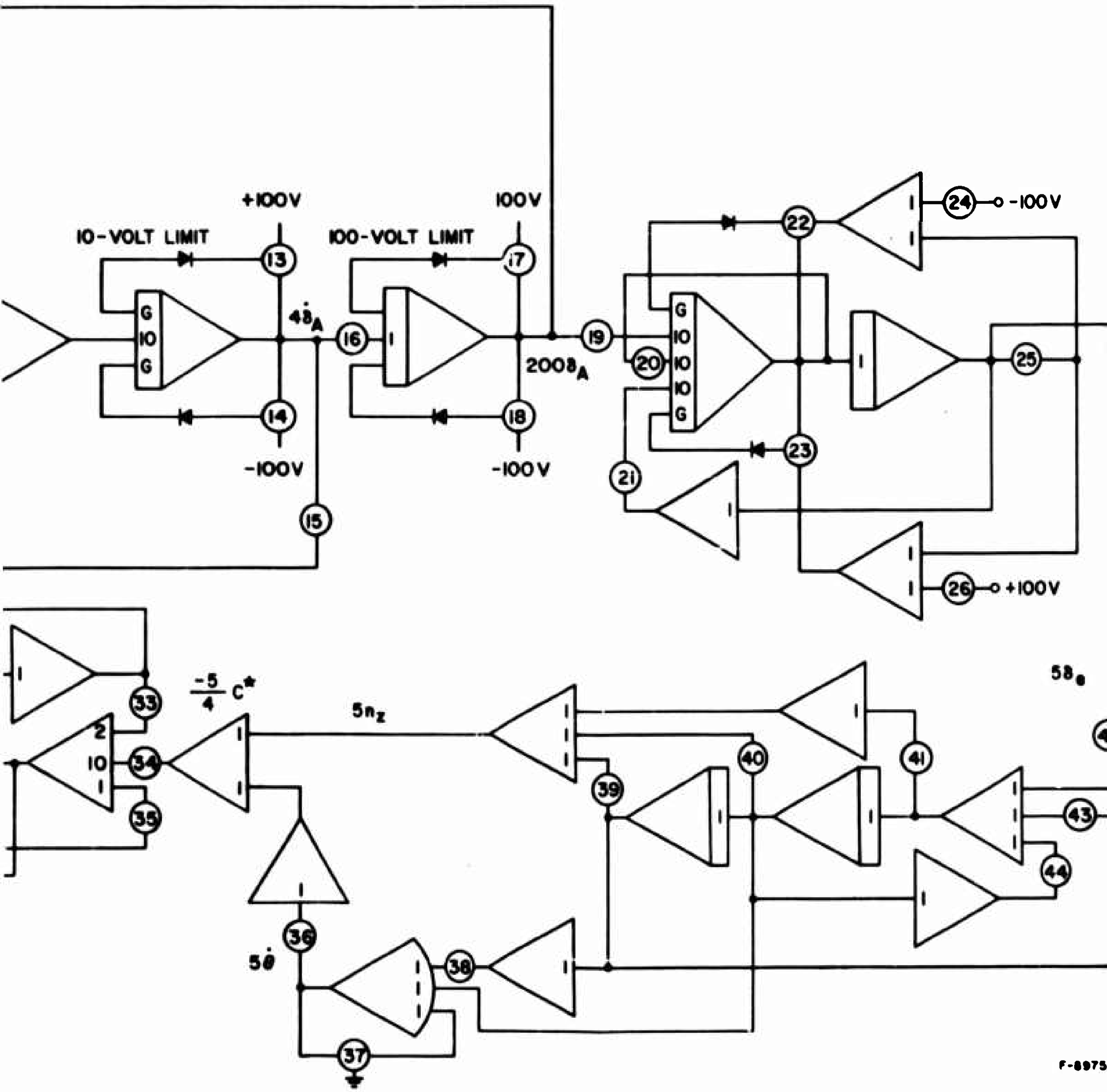


Figure 69
Fail-Passive Actuator Concept
Analog Diagram

TABLE VII

FAIL-PASSIVE ACTUATOR SIMULATION POTENTIOMETER SETTINGS

Pot Number	Setting	Function	Pot Number	Setting	Function
1	1.0	Channel Match	27	0.2	Forward
2	1.0		28	0.2	Model Plus
3	1.0		29	Variable	Input Command
4	0.068	Servo Amplifier	30	0.8	
5	0.068	Current Limits	31	0.64	Inverse
6	0.068		32	0.16	Model
7	0.068		33	0.85	Plus Shaping
8	0.068		34	0.1875	
9	0.068		35	0.3	
10	0.24	Force Output of	36A	0.25 HQ	
11	0.24	Valves	36B	0.25 LQ	
12	0.24		37A	0.0725 HQ	
13	0.5	Rate Limit	37B	1.0 LQ	Airframe
14	0.5	Secondary Actuator	38A	0.1333 HQ	Dynamics
15	0.333	Flow Feedback	38B	0.0305 LQ	
16	0.5	Secondary	39A	0.7679 HQ	HQ = High
17	0.5	Actuator Position	39B	0.0003 LQ	Dynamic
18	0.5	Limit	40A	0.1616 HQ	Pressure
19	0.4	Actuator/Elevator	40B	0.022 LQ	
20	0.2	Model	41A	0.64 HQ	LQ = Low
21	0.4		41B	0.047 LQ	Dynamic
22	0.5	Elevator	42A	0.282 HQ	Pressure
23	0.5	Rate	42B	0.127 LQ	
24	0.2	Limits	43A	0.282 HQ	
25	0.4		43B	0.0048 LQ	
26	0.2		44A	0.4284 HQ	
			44B	0.0753 LQ	

The following limits were imposed on the simulated actuator: servo amplifier (40 ma), the intermediate actuator rate (2.5 inches/second), intermediate actuator position (0.5 inch), and surface actuator rate (40 degrees/second).

The following assumptions were made in the development of the simulation for the fail-passive actuator mechanization.

a. Triplex hydraulics and quadruplex electrical supplies.

b. The artificial feel sensors are a nonredundant unit.

c. The scheduled gain G is constant for the flight condition under investigation but is varied for different flight conditions to maximize loop gain.

d. The demodulators, modulators, amplifiers and torquers are linear in the frequency range of interest.

e. Linear two-degree-of-freedom airframe dynamics for a typical high performance aircraft are adequate for determining response characteristics.

Time scaling by a factor of 10 is employed. The hydraulic power to a failed channel is not shut off in the simulation, resulting in a one-third loss in dynamic response of the intermediate actuator per failed channel. An improvement in dynamic response of the intermediate actuator results if the hydraulic power to a failed channel is shut off. A thorough discussion of this actuator design and its failure modes was presented in Section VI. The monitors are not simulated since failure reporting or switching is not done.

Only passive failures are investigated in the simulation. The transient and performance degradation for active failures are very similar to that which results for passive failures. Further, the relative probability of an active failure is only 0.15 percent. Figures 70, 71 and 72 present time histories of selected aircraft variables in response to a step stick displacement. Channels 1 and 2 show C^*_c and the resulting C^*. Elevator rate and position are shown on channels 3 and 4. Channels 5 and 6 show aircraft normal acceleration and pitch attitude rate. The aircraft is at the low dynamic pressure flight condition.

Figure 70 demonstrates the operation of the fail-passive actuator in response to a step stick displacement with no failures. It can clearly be observed in this figure that no airframe transients result from the failures which occur at $t = 3$ and $t = 7$ seconds. This result bears out the theory and laboratory test on the system for a static condition. Figures 71 and 72 are step responses under conditions of one and two failures respectively. A comparison of these responses and those of figure 70 show that the dynamic performance of the actuator did not observably degrade the aircraft response to any degree. Figures 73 and 74 show time histories of the fail-passive actuator scheme when responding to 1 hertz (which corresponds to very fast command signals) and 4 hertz (which corresponds to a stability augmentation signal frequency) input signals respectively. At 1-hertz input signal, essentially no degradation in performance could be observed under the failed conditions. However, the elevator rate at the 4-hertz input showed about 10 percent degradation in magnitude with one failure and almost 50 percent degradation with two failures. As the command frequency is increased, the percentage degradation will also increase.

Figures 75 and 79 show the time responses of the fail-passive actuator at the high dynamic pressure flight condition. A comparison of figures 75, 76 and 77 shows essentially no change in dynamic performance of the aircraft with two channel failures. Again as at the low dynamic pressure region, failures which occur when the system is in a static

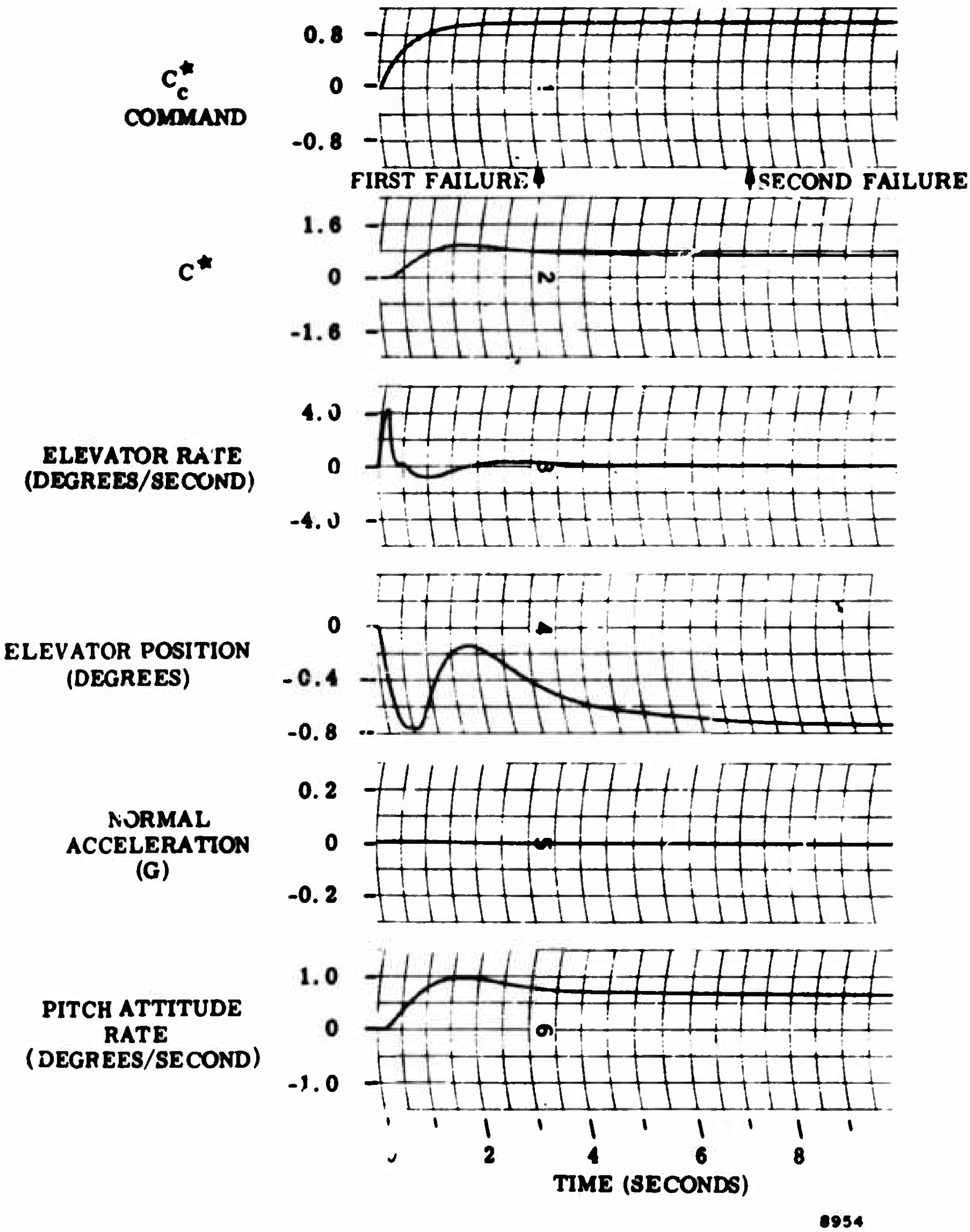


Figure 70
Fail-Passive Actuator at Low Dynamic Pressure

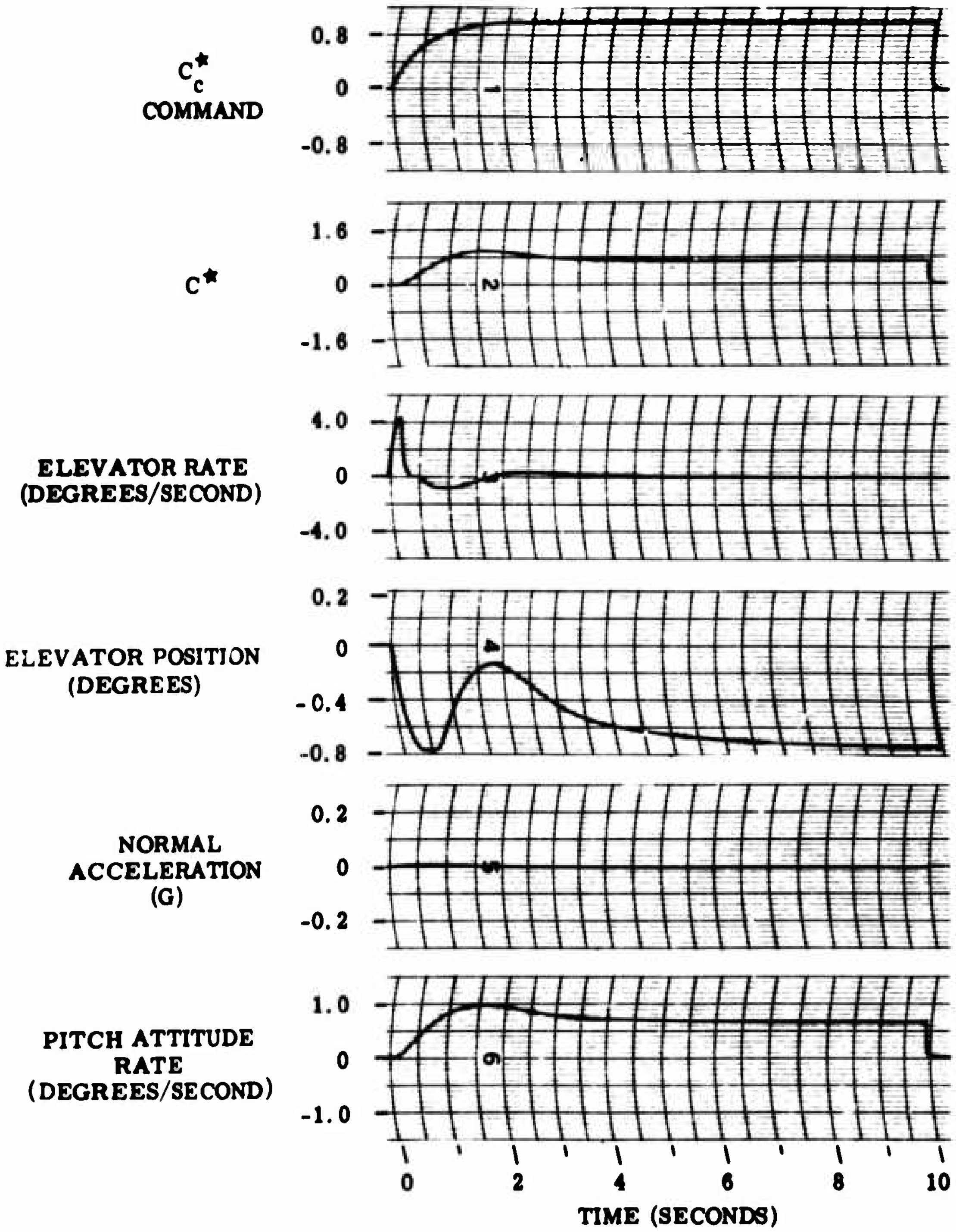


Figure 71
Fail-Passive Actuator at Low Dynamic Pressure One Failure

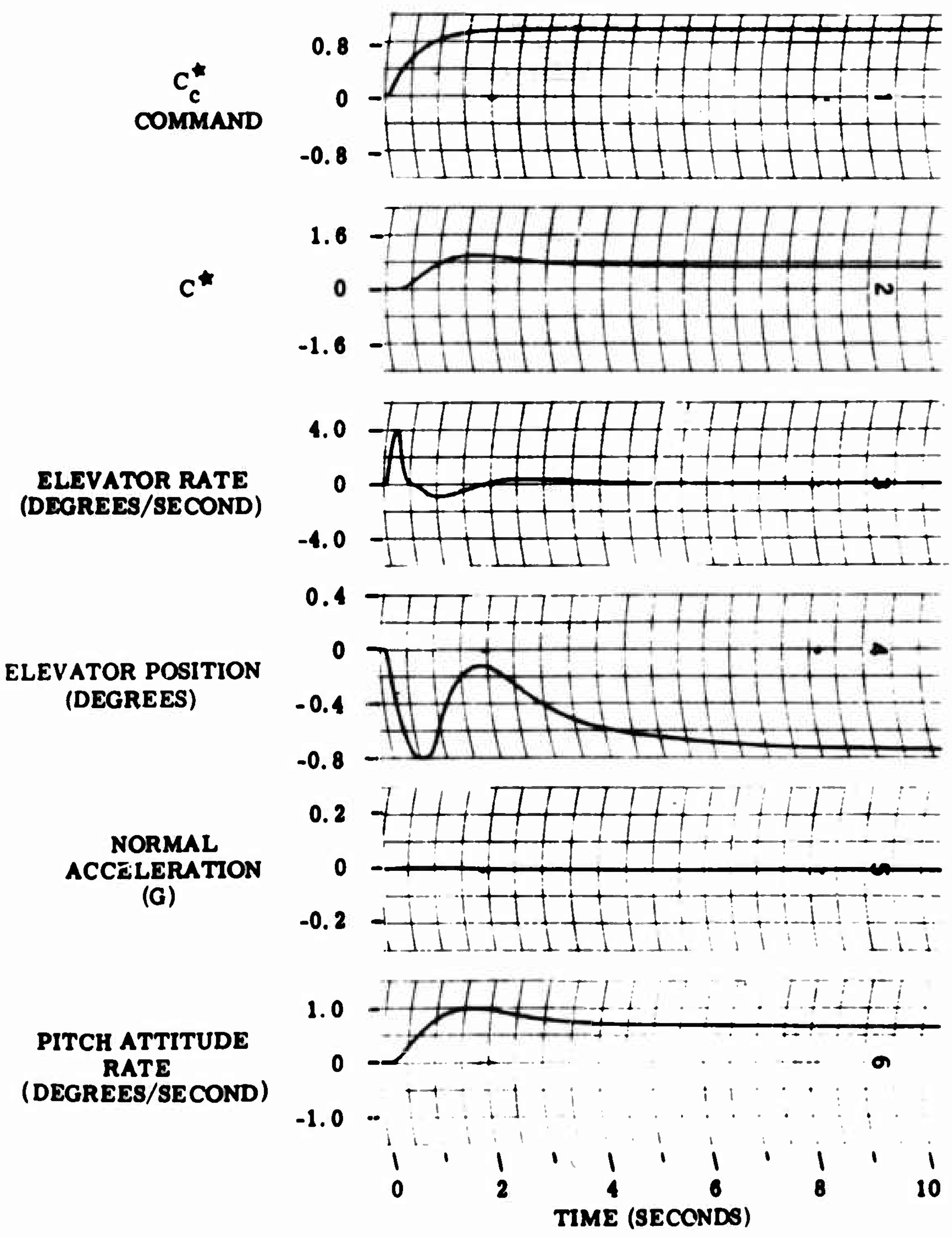


Figure 72
Fail-Passive Actuator at Low Dynamic Pressure Two Channels Failed

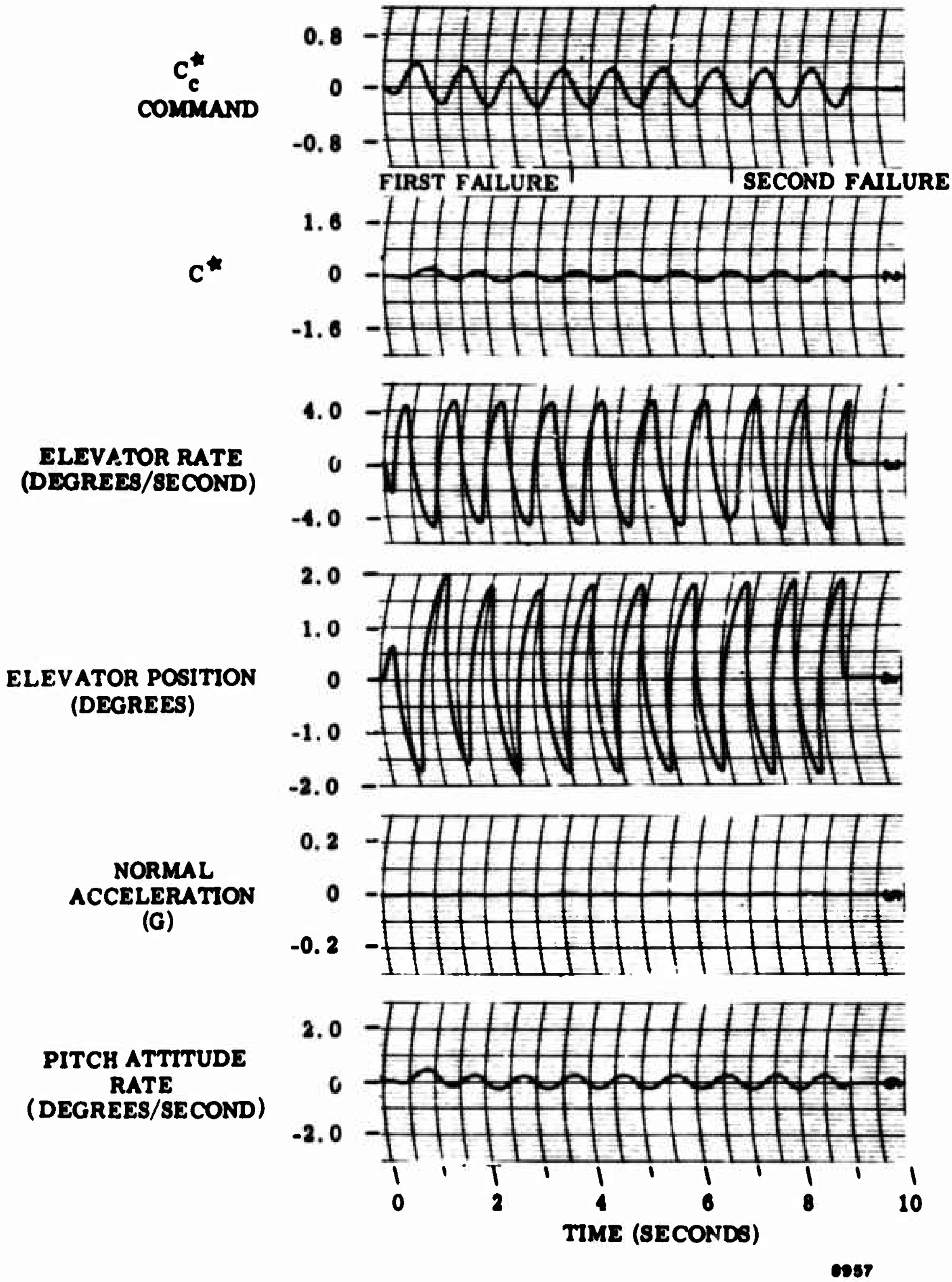


Figure 73
Fail-Passive Actuator at Low Dynamic Pressure 1-Hertz Input

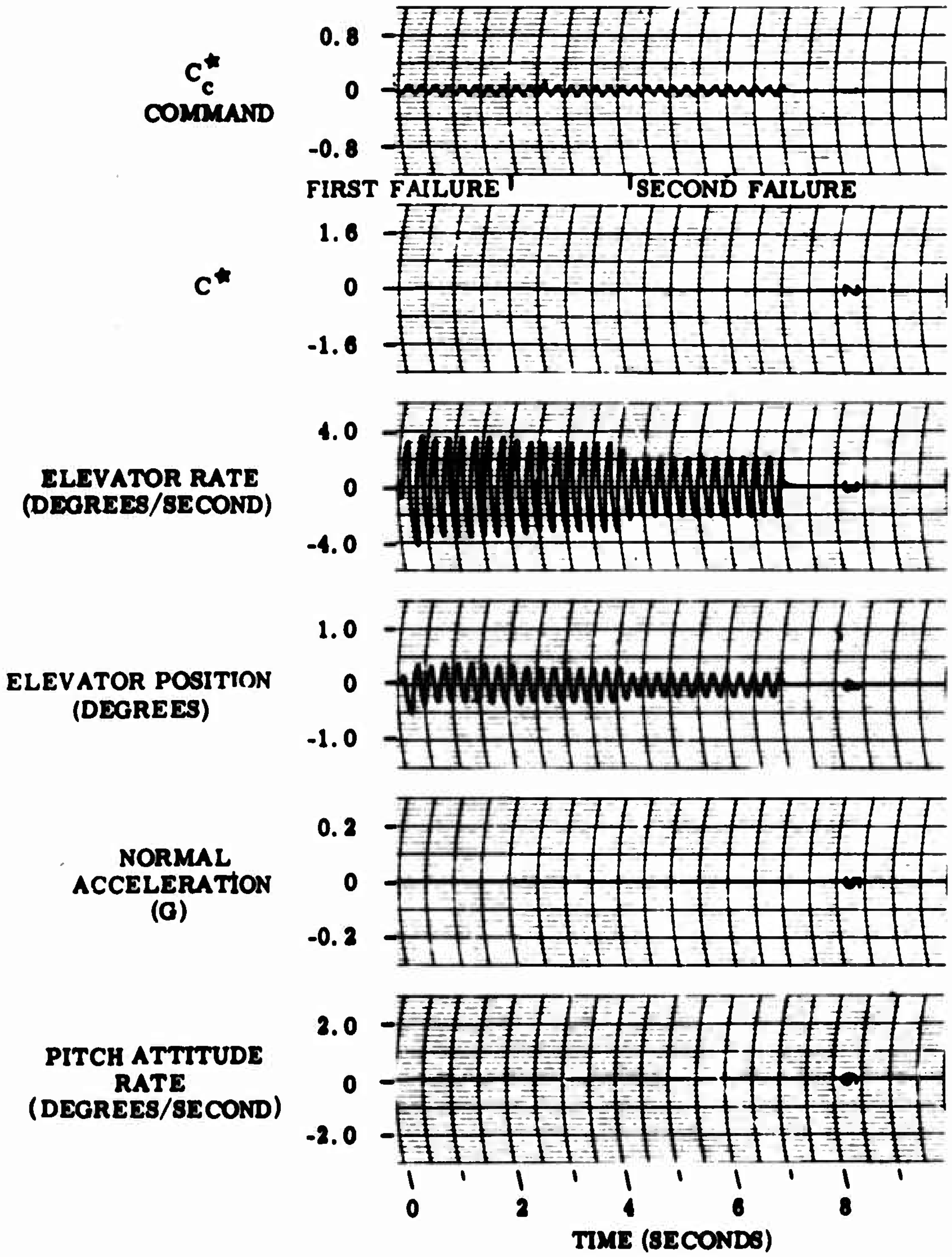


Figure 74
Fail-Passive Actuator at Low Dynamic Pressure 4-Hertz Input

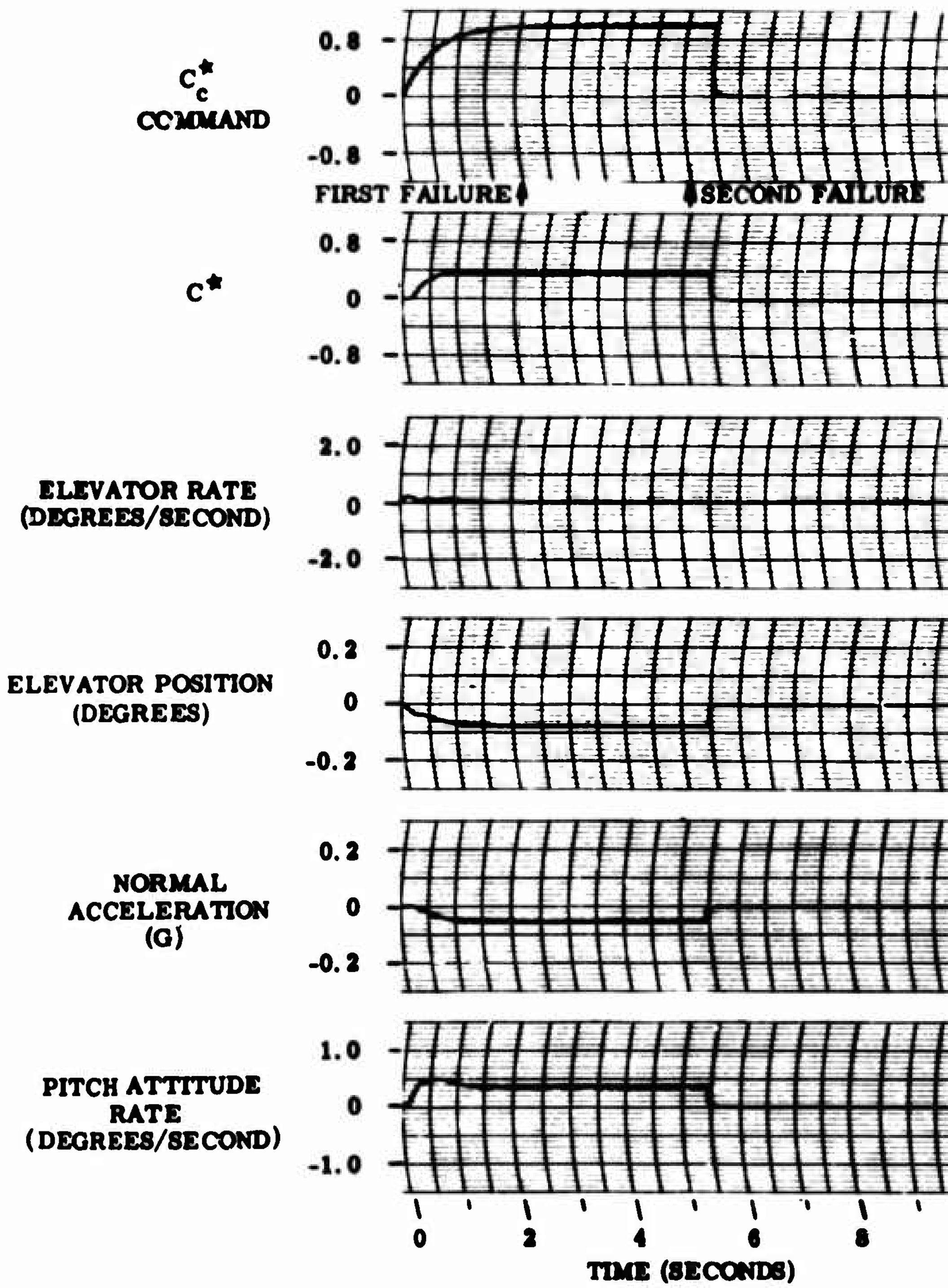


Figure 75
Fail-Passive Actuator at High Dynamic Pressure

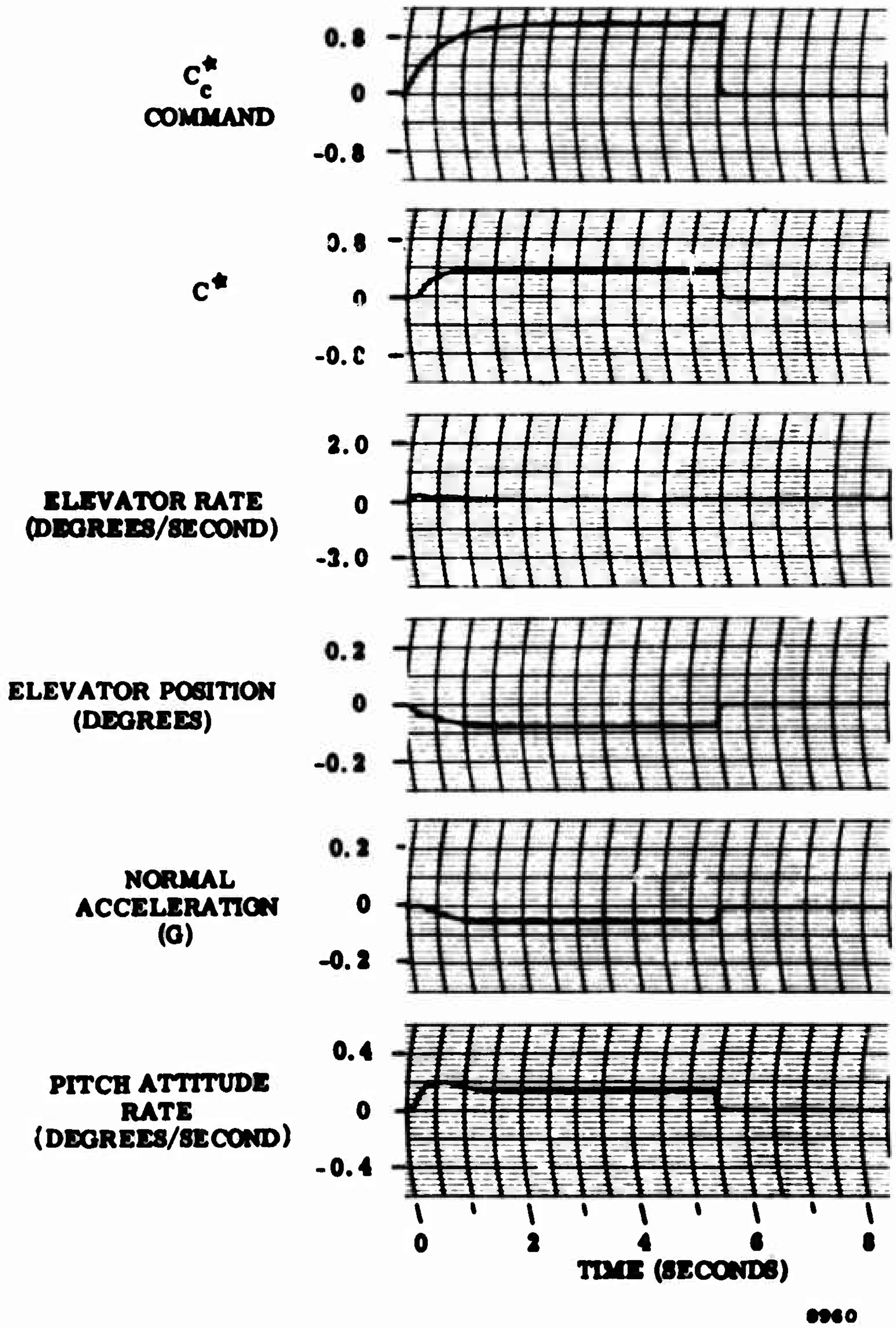


Figure 76
Fail-Passive Actuator at High Dynamic Pressure One Failure

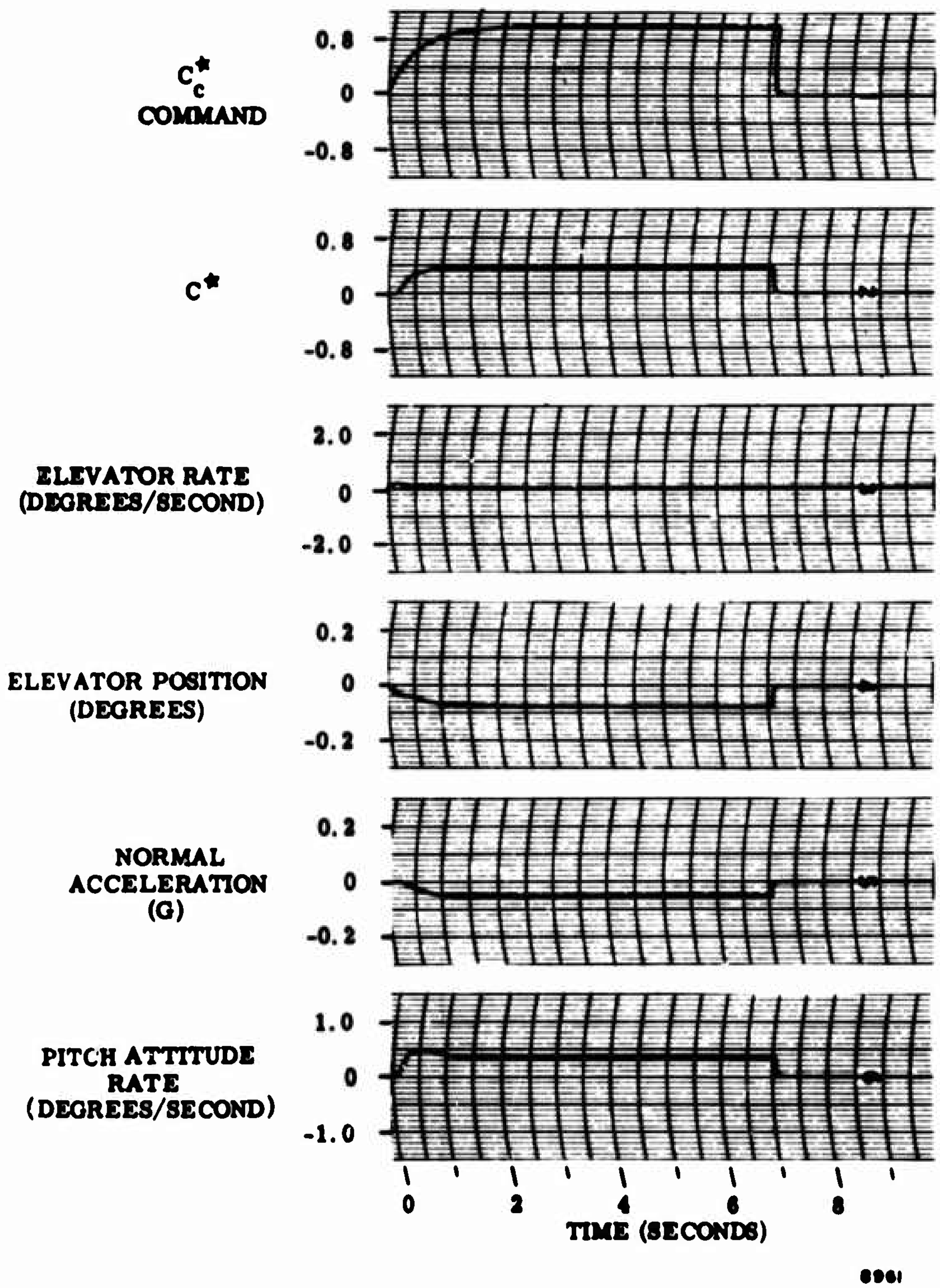


Figure 77
Fail-Passive Actuator at High Dynamic Pressure Two Channels Failed

condition result in no airframe transients. Figures 78 and 79 show time response for 1-hertz and 4-hertz command signals. These results are similar to those at the low dynamic pressure region in that no degradation is evident at 1 hertz while 50 percent degradation in surface rate results from two failures at 4 hertz.

The results of the simulation demonstrate that the fail-passive actuator meets all of the requirements for a fly-by-wire system that remains operational after double failures. No system performance degradation occurs in the normal pilot control frequency range, but up to 50 percent degradation occurs at the augmentation frequencies. The requirement for remaining operational after double failures refers to the flight control system not to the actuator. While the fail-passive actuator does become degraded somewhat due to failures, system performance is not noticeably affected. This result is one of the features of the C* command system as discussed in Section III. A loss of hydraulic power was not simulated. Presumably such a failure would have a more noticeable affect on performance where active redundancy is employed than would a control channel failure. Another factor to consider is that a very high performance aircraft such as the F-111 has been simulated. Degradation would be less noticeable on a lower performance aircraft.

The fail-passive actuator has several advantages which far overshadow the slight degradation in performance with failures. For fly-by-wire application, three active channels provide a two-fail-operational system with no requirements for switching. The fail-passive actuator design can include switching for pilot channel selection and for improving system performance upon failure indication, but these switches are not necessary for two-fail-operational capability. Additional benefits are derived from the system capability to withstand failures without causing an aircraft transient.

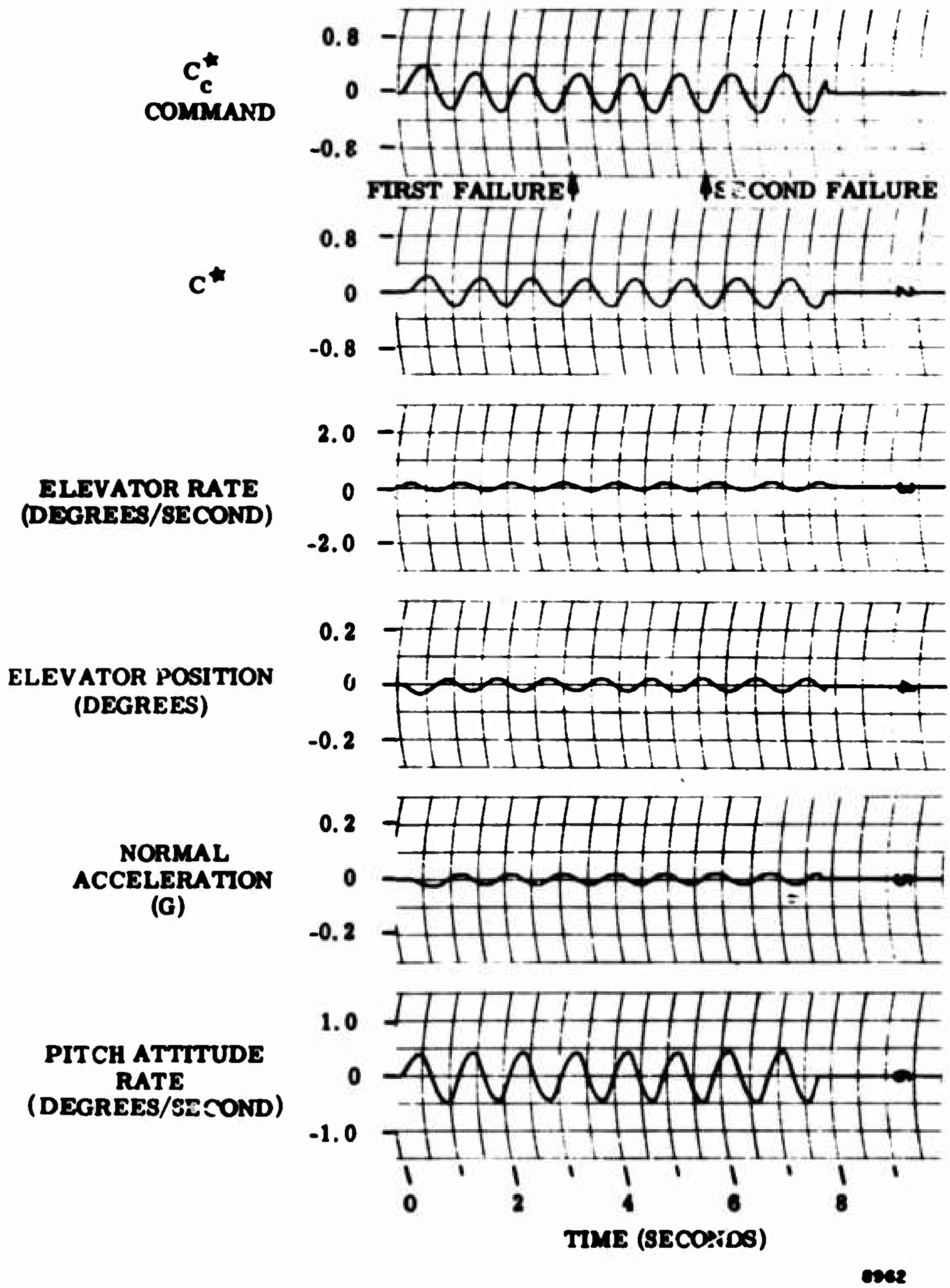


Figure 78
Fail-Passive Actuator at High Dynamic Pressure 1-Hertz Input

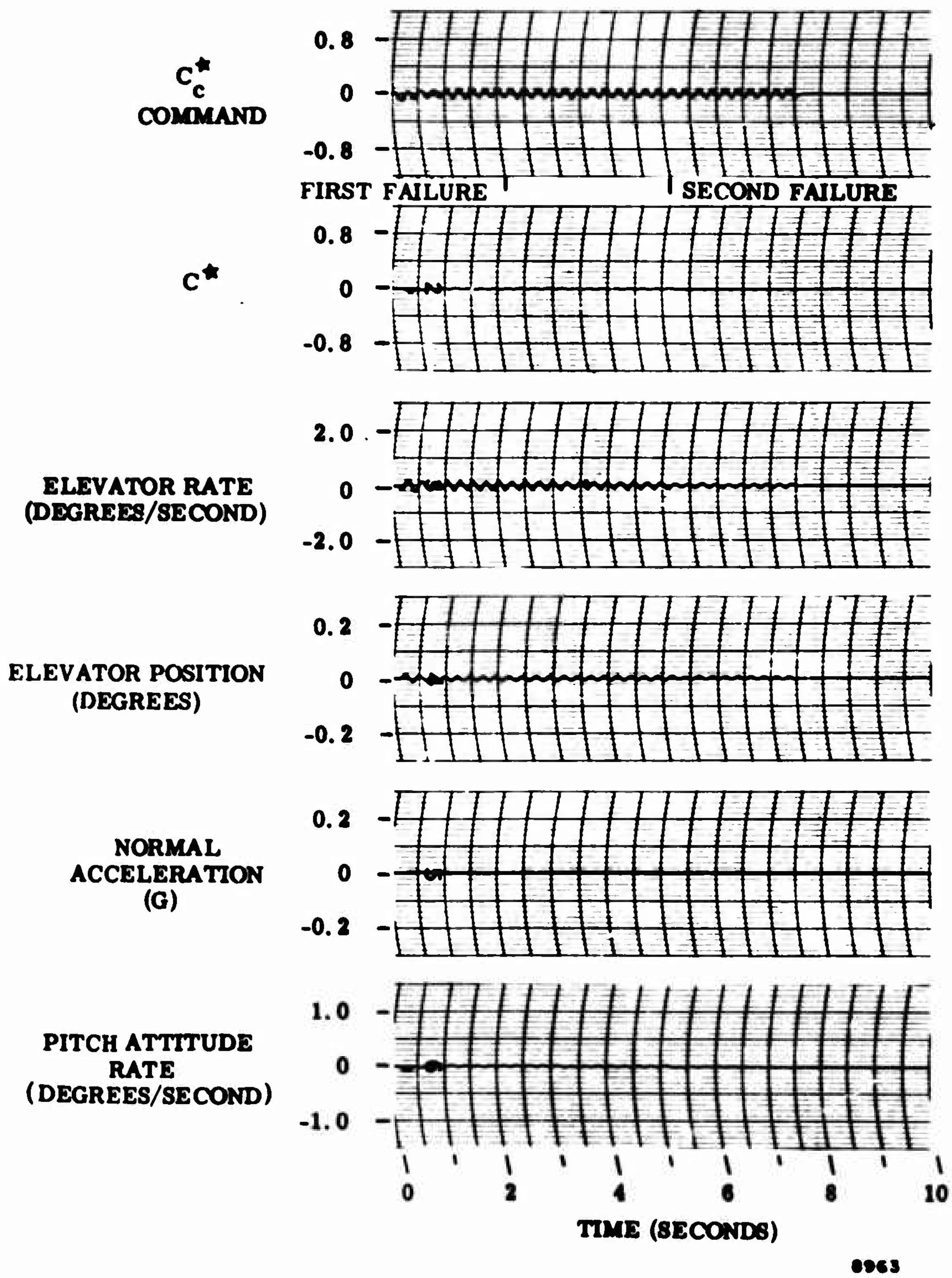


Figure 79
Fail-Passive Actuator at High Dynamic Pressure 4-Hertz Input

SECTION VIII

COMPARISON OF MECHANICAL AND FLY-BY-WIRE SYSTEMS

Three aircraft, the B-52H, F-111, and CH-46, were chosen as test aircraft to compare their mechanical control systems with equivalent fly-by-wire systems. These aircraft represent a cross section of the various classes of aircraft in service today, that is, the heavy bomber or transport, the high performance fighter/interceptor, and the VTOL aircraft. The comparison is based on cost, weight and space only for the B-52H and F-111, since reliability and maintainability data are not readily available. Weight and cost of the structural parts were determined from their volume, estimated total aircraft weight, and estimated total aircraft cost. The comparison for the CH-46 is based on cost, weight, reliability, and maintainability. The volume of the existing control system is not known.

$$\text{B-52H} \left\{ \begin{array}{ll} \text{Cost/cubic inch} & = 25.00 \, \dfrac{\text{dollars}}{\text{pound}} \times 0.1219 \, \dfrac{\text{pound}}{\text{cubic inch}} \\ & = 3.04 \text{ dollars/cubic inch} \\ \text{Pounds/cubic inch} & = (1.25)\ (0.0925) = 0.1219 \, \dfrac{\text{pound}}{\text{cubic inch}} \end{array} \right.$$

$$\text{F-111} \left\{ \begin{array}{ll} \text{Cost/cubic inch} & = 35.70 \, \dfrac{\text{dollars}}{\text{pound}} \times 0.1219 \, \dfrac{\text{pound}}{\text{cubic inch}} \\ & = 4.35 \text{ dollars/cubic inch} \\ \text{Pounds/cubic inch} & = (1.25)\ (0.0925) = 0.1219 \, \dfrac{\text{pound}}{\text{cubic inch}} \end{array} \right.$$

1. B-52H FLIGHT CONTROL SYSTEM

The elevator control system on the B-52H aircraft, figure 80, is a manually operated cable system employing two aerodynamically and statically balanced, tab operated, floating surfaces. Control movements are transmitted from the pilot's or copilot's control column through column disconnect mechanisms to their respective control cable quadrants. The independent control cables are bussed together fore and aft at the control cable quadrants and torque tube mounted tension regulators respectively. Also attached to the torque tube are autopilot servo quadrant, q-spring, and control tab linkage. Control tabs, limited in travel to ±20 degrees by gust dampers, are located on each independently hinged elevator half and are mechanically bussed together by the control tab linkage. This technique results in identical motion of the elevator halves with low pilot effort. The q-spring produces a "feel" force as a function of indicated airspeed to eliminate over-control of the aircraft by the pilot.

A list of control system basic components used to determine the weight, space and equivalent costs for the B-52H elevator control system follows:

- Pilot, copilot control column
- Column disconnect
- Cable control quadrant
- Cable (3/16 inch diameter)
- Turnbuckle
- Pushrods (5/8 inch diameter)
- Pulleys
- Elevator torque tube
- Cable tension regulator
- AFCS cable quadrant
- Gust damper

Figure 81 shows a fly-by-wire equivalent of the B-52H elevator control system. Quadruplex position transducers on the control column and elevator torque tube provide command and feedback signals respectively. Trim, artificial feel and AFCS signals terminate in the control electronics unit. The entire system is quadruplex with operation set up for dual power sources. The fly-by-wire equivalent replaces the mechanical system only between the control column and the torque tube to minimize installation problems associated with placing an actuator in a position to control motions of a control tab.

Table VIII presents the weight, volume and cost figure for each of the components of the B-52 elevator control system and its fly-by-wire equivalent. The fly-by-wire equivalent costs approximately twice as much as the mechanical system, but results in a weight savings of 125 pounds and a space savings of 865 cubic inches.

The spoiler control system on the B-52H aircraft, figures 82 and 83, is a manually operated system with control motion originating at either the pilot's or copilot's control wheel. Control wheels are bussed together by cables at both the control wheel drums and the rear spar drums. Control motions are applied to metering valves which cause the 14 spoiler actuators to extend or retract. On each wing, the four outboard spoilers are linked together and called group A; the three inboard spoilers are linked together and called group B. Only one followup linkage (feedback member) is used for each group. Lateral trim and AFCS electrical signal drive trim actuators and AFCS servo drums respectively which are connected to the spoiler actuate metering valves through cables. The lateral mechanical control system is made up of the following principal components:

- Pilot and copilot control wheel
- Control wheel drum
- Cables
- Turnbuckles
- Pulleys
- Rear spar drum (lower and upper)
- Servo motor drum (AFCS drum)
- Pushrods
- Tension regulator

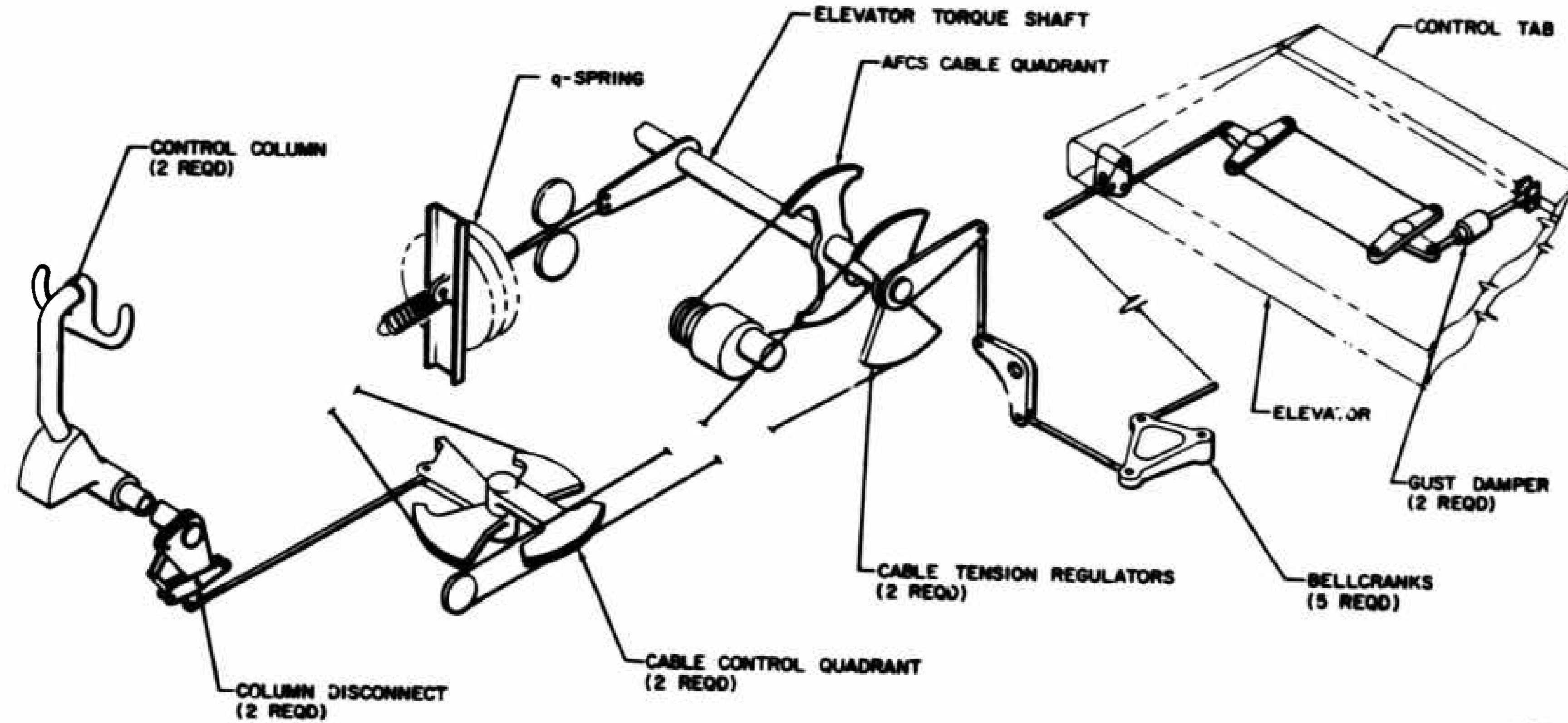


Figure 80
B-52H Elevator Control System (Mechanical)

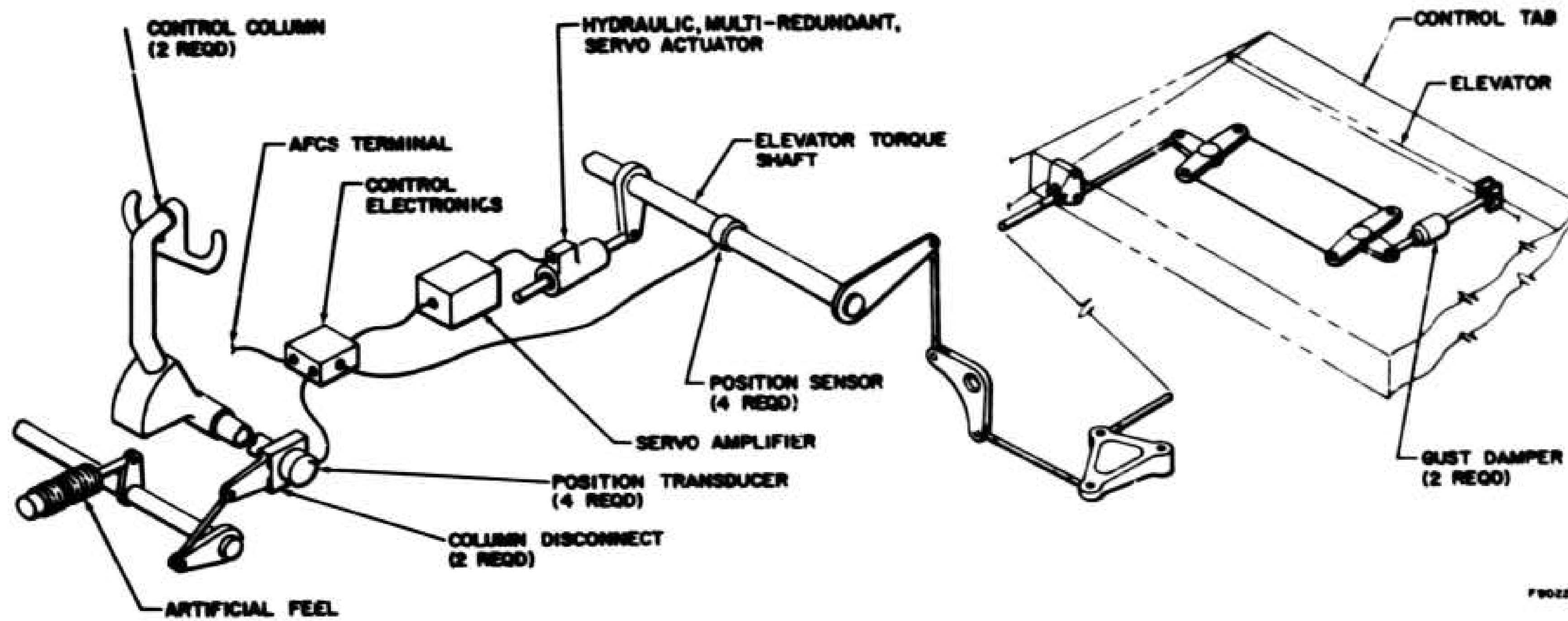


Figure 81
B-52H Elevator Control System (Fly-By-Wire)

TABLE VIII

B-52H ELEVATOR CONTROL SYSTEM WEIGHT AND CO

Control System Component	Individual			Mechanical Syste		
	Weight (lb)	Volume ($in.^3$)	Cost (dollars)	Number	Weight (lb)	Volu (in.
Control Column	35.4	291	885	2	70.8	582
Control Disengage	4.6	37.7	115.0	2	9.2	75.4
Cable Control Quadrant	22.5	184	570.0	2	45	368
(Cable 3/16 in. Diameter)	0.084/ft	0.332/ft	0.505/ft	475 ft	39.9	157.
Pulleys (4 to 6 in. Diameter)	0.192 lb ea	3.14 ea	10.0 ea	48	9.216	150.
Turnbuckles	0.279 lb ea	1.15 ea	7.0	28	7.81	32.2
Pushrods (5/8 in. Diameter, 1/8 in. Wall)	0.161 lb/ft	1.32 ft	4.0	70 ft	11.27	92.4
Elevator Torque Shaft	12.7	104	316.0	1	12.7	104
Tension Regulators	13.8	113	354.0	2	27.6	226.
AFCS Quadrant	17.5	144	438.	1	17.5	144
Gust Dampers	2.9	24.5	74.5	2	5.8	49.0
Bellcranks	2.45 ea	20 ea	60.8 ea	10	24.5	200
Position Transducer	0.37 ea	10	100.0	---	---	---
Control Electronics Channel Servo Amplifier/Stage	1.25 ea	20 ea	2000	---	---	---
Surface Actuator*	7.2	160	5000.0	---	---	---
Electronics Transmission (#22) (Shielded)	0.023/ft	0.094/ft	0.10/ft	---	---	---
Force Feel Spring	7.62	62.6	191.0	---	---	---
AFCS Servo	5.80	47.7	1500	1	5.80	47.7
Total System Characteristics	---	---	---	---	287.10	2228

*Redundant Hydraulic Actuator (Low Force)

TABLE VIII

-52H ELEVATOR CONTROL SYSTEM WEIGHT AND COST

al		Mechanical System				Fly-By-Wire System			
	Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars
	885	2	70.8	582	1770	2	70.8	582	1770
	115.0	2	9.2	75.4	230.0	2	9.2	75.4	230.0
	570.0	2	45	368	1140.0	---	---	---	---
ft	0.505/ft	475 ft	39.9	157.7	239.87	20 ft	1.68	6.64	10.1
a	10.0 ea	48	9.216	150.72	480	4	0.768	12.56	40
a	7.0	28	7.81	32.2	196	2	0.558	2.30	14
t	4.0	70 ft	11.27	92.4	280	35 ft	5.64	46.2	140
	316.0	1	12.7	104	316.0	1	12.7	104	316.0
	354.0	2	27.6	226.0	708	---	---	---	---
	438.	1	17.5	144	438	---	---	---	---
	74.5	2	5.8	49.0	149.0	2	5.8	49.0	149.0
	60.8 ea	10	24.5	200	608	8	19.60	160	486.40
	100.0	---	---	---	---	8	1.0	10.0	8000
	2000	---	---	---	---	4	5	80	---
	5000.0	---	---	---	---	1*	7.2	0.60	5000.0
/ft	0.10/ft	---	---	---	---	570	13.10	53.58	57.0
	191.0	---	---	---	---	1	7.62	62.6	191.0
	1500	1	5.80	47.7	1500	---	---	---	---
	---	---	287.10	2228.8	8054.87	---	160.66	1363.82	17,203.5

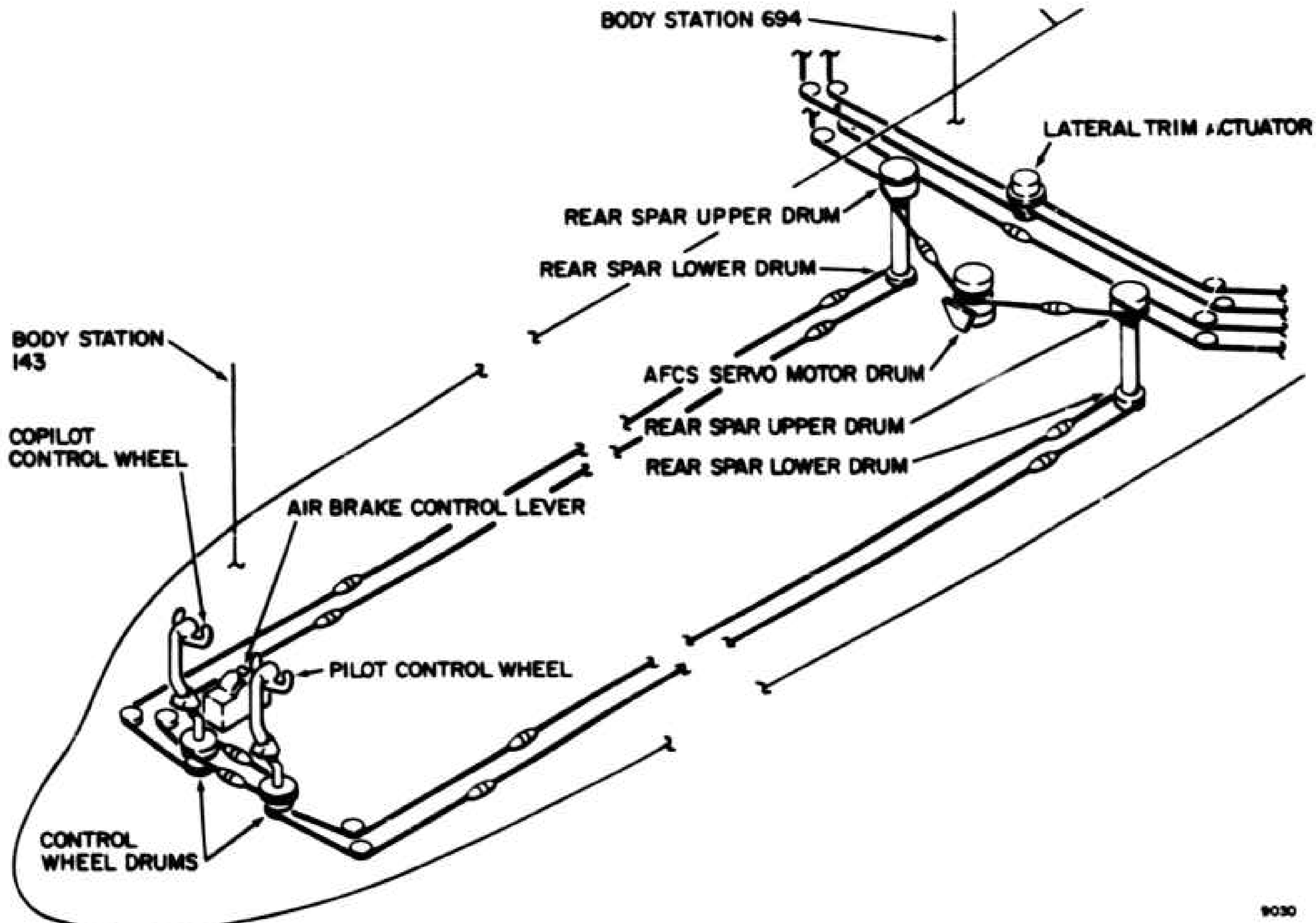


Figure 82
B-52H Roll Control System (Mechanical)

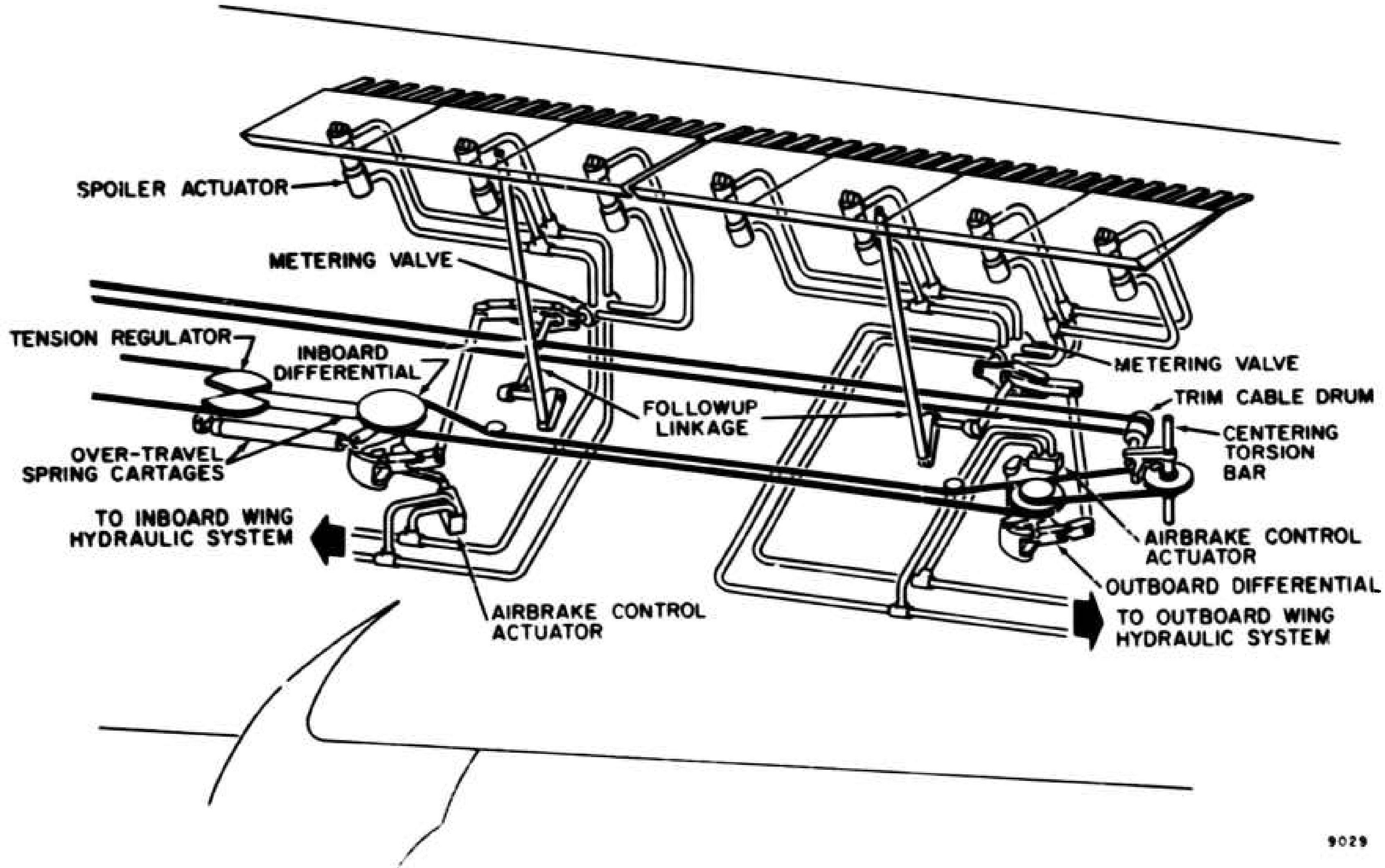


Figure 83
B-52H Spoiler System (Mechanical)

Inboard differential
Outboard differential
Trim cable drum
Centering torsion bar
Metering valve
Spoiler actuators

Figure 84 shows the fly-by-wire equivalent of the mechanical B-52H spoiler control system. Quadruplex position transducers at the control wheel provide roll command signals to the servovalves which drive the spoiler actuators. Mechanical feedback to the valves is employed as in the original system. Trim, AFCS, airbrake and manual control signals are summed at the control electronics.

Table IX presents the weight, volume, and cost of the original B-52H spoiler system and for the fly-by-wire equivalent. The fly-by-wire equivalent costs approximately twice as much as the mechanical system, but it results in a weight savings of 155 pounds and a space savings of 1195 cubic inches.

The rudder control system on the B-52H, figure 85, is a manually operated cable system employing an aerodynamically and statically balanced, tab operated surface. Control movements are generated by operation of the rudder pedals from either the pilot's or copilot's station. The left and right hand rudder pedals are bussed together fore and aft at the control quadrants and torque tube mounted tension regulators respectively. Also attached to the torque tube are autopilot servo, q-spring for artificial feel, and rudder trim. Control motions are transmitted to the rudder control tab by additional bellcranks and linkages. An additional tab (stabilizer tab) is picked up when large deflections are commanded. The mechanical rudder system is made up of the following list of major components:

Pilot and copilot rudder pedals
Rudder control cable quadrants
Cables (3/16 inch)
Pushrods
Turnbuckles
Torque tube
Cable tension regulator
AFCS servo quadrant
Trim actuator
Gust damper
Aft torsion bar

Figure 86 shows a fly-by-wire equivalent of the B-52H rudder control system. The system employs quadruplex position transducers for yaw command signals and actuator feedback signals. Quadruplex electronics and servo amplifiers are used to provide two-failure operation. AFCS and trim signals electrically sum with the command signal in the control electronics, but the stability augmentation and gust damper remain mechanical. The fly-by-wire equivalent replaces all of the mechanical linkages between the rudder pedals and the control linkages in the vertical fin.

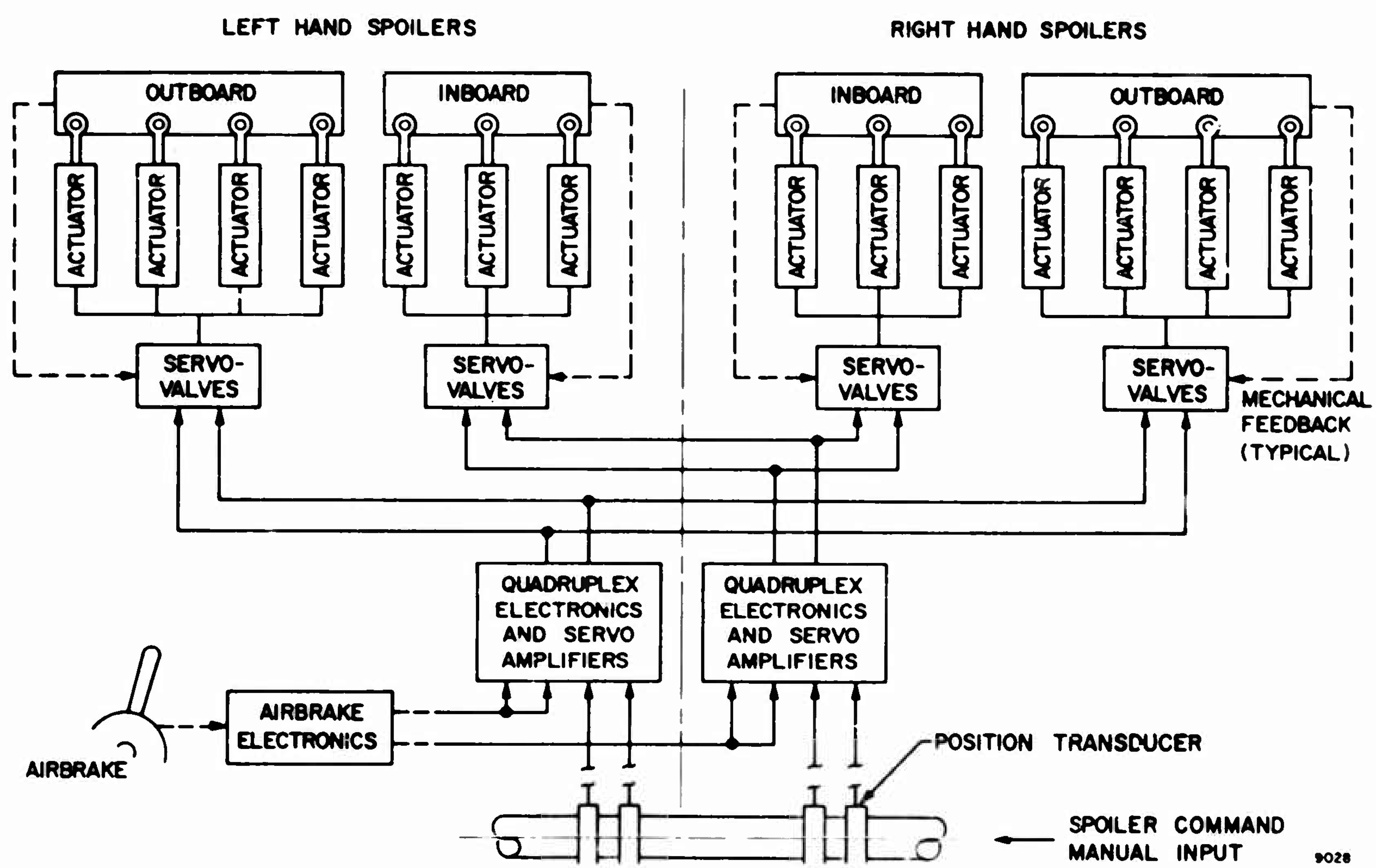


Figure 84
B-52H Roll Control System (Fly-By-Wire)

TABLE IX

B-52H ROLL CONTROL SYSTEM WEIGHT AND CO

Control System Component	Individual			Mechanical Sy		
	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	V (
Control Wheel	24.9	204.9	621.0	2	49.8	4
Control Wheel Drum	9.79	77.2	235.0	2	19.58	1
Cable (3/16 in. Diameter)	0.084/ft	0.332/ft	0.505/ft	360 ft	30.24	1
Turnbuckles	0.279 ea	1.15 ea	7.00 ea	44	12.276	5
Pulleys (4 to 6 in. Diameter)	0.192	3.14 ea	10.00 ea	32	6.144	1
Rear Spar Drum	9.18	75.5	229.0	2	18.36	1
AFCS "Servo" Drum	0.956	7.85	23.9	1	0.956	7
Pushrods (5/8 in. Tube)	0.161/ft	1.32/ft	4.0/ft	36 ft	5.796	4
Cable Tension Regulator	4.81	39.3	119.0	2	9.62	7
INBD Differential	12.8	105.4	319.0	2	25.6	2
OUTD Differential	9.65	78.1	241.0	2	19.3	1
Trim Servo Drum	0.956	7.85	23.9	1	0.956	7
Centering Torsion Bar	3.81	31.32	95.0	2	7.62	6
Spoiler and Airbrake Metering Valve	3.41	42.0	780.0	8	27.28	3
Spoiler Actuators	1.94	15.96	300.0	14	27.16	2
Over Travel Cart	7.90	64.8	197.0	4	31.6	2
Position Transducer	0.37 ea	10	100.0	---	---	-
Control Electronic/Channel Servo Amplifier	1.25 lb	20	2000.0	---	---	-
Control Valve*	2.42	24.0	2500	---	---	-

*Redundant (Hydraulic) Servo.

TABLE IX

²H ROLL CONTROL SYSTEM WEIGHT AND COST

	Mechanical System				Fly-By-Wire System			
Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)
621.0	2	49.8	409.8	1242.0	2	49.8	409.8	1242.0
235.0	2	19.58	154.4	470.0	2	19.58	154.4	470.0
0.505/ft	360 ft	30.24	119.52	181.8	10 ft	0.84	3.32	5.05
7.00 ea	44	12.276	50.6	308.0	2	0.578	2.30	14.0
10.00 ea	32	6.144	100.48	320.0	2	0.384	6.28	20.0
229.0	2	18.36	151.0	458.0	---	---	---	---
23.9	1	0.956	7.85	23.9	---	---	---	---
4.0/ft	36 ft	5.796	47.52	144.0	---	---	---	---
119.0	2	9.62	78.6	238.0	---	---	---	---
319.0	2	25.6	210.8	638.0	---	---	---	---
241.0	2	19.3	156.2	482.0	---	---	---	---
23.9	1	0.956	7.85	23.9	1	0.956	7.85	23.9
95.0	2	7.62	62.64	190.0	---	---	---	---
780.0	8	27.28	336.0	6240	---	---	---	---
300.0	14	27.16	223.44	4200	14	27.16	223.44	4200
197.0	4	31.6	259.2	788.0	---	---	---	---
100.0	---	---	---	---	4	1.48	40.0	400.0
2000.0	---	---	---	---	8	10.0	160.0	16000
2500	---	---	---	---	4	9.68	96.0	10,000

TABLE IX (cont)

B-52H ROLL CONTROL SYSTEM WEIGHT AND CO

Control System Component	Individual			Mechanical Sy		
	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	V (i
Force Feel Spring	7.62	62.64	191.0	---	---	-
Electrical (Shielded) Cable	0.023/ft	.094/ft	0.10/ft	---	---	-
AFCS Servo	5.80	41.7	1500	1	5.80	4
Total System Characteristics	---	---	---	---	298.1	2

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TABLE IX (cont)

2H ROLL CONTROL SYSTEM WEIGHT AND COST

	Mechanical System				Fly-By-Wire System			
Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)
191.0	---	---	---	---	1	7.62	62.64	191.0
0.10/ft	---	---	---	---	664 ft	15.27	62.4	66.40
1500	1	5.80	47.7	1500.0	---	---	---	---
---	---	298.1	2423.6	17,447.6	---	143.34	1228.4	32,632.5

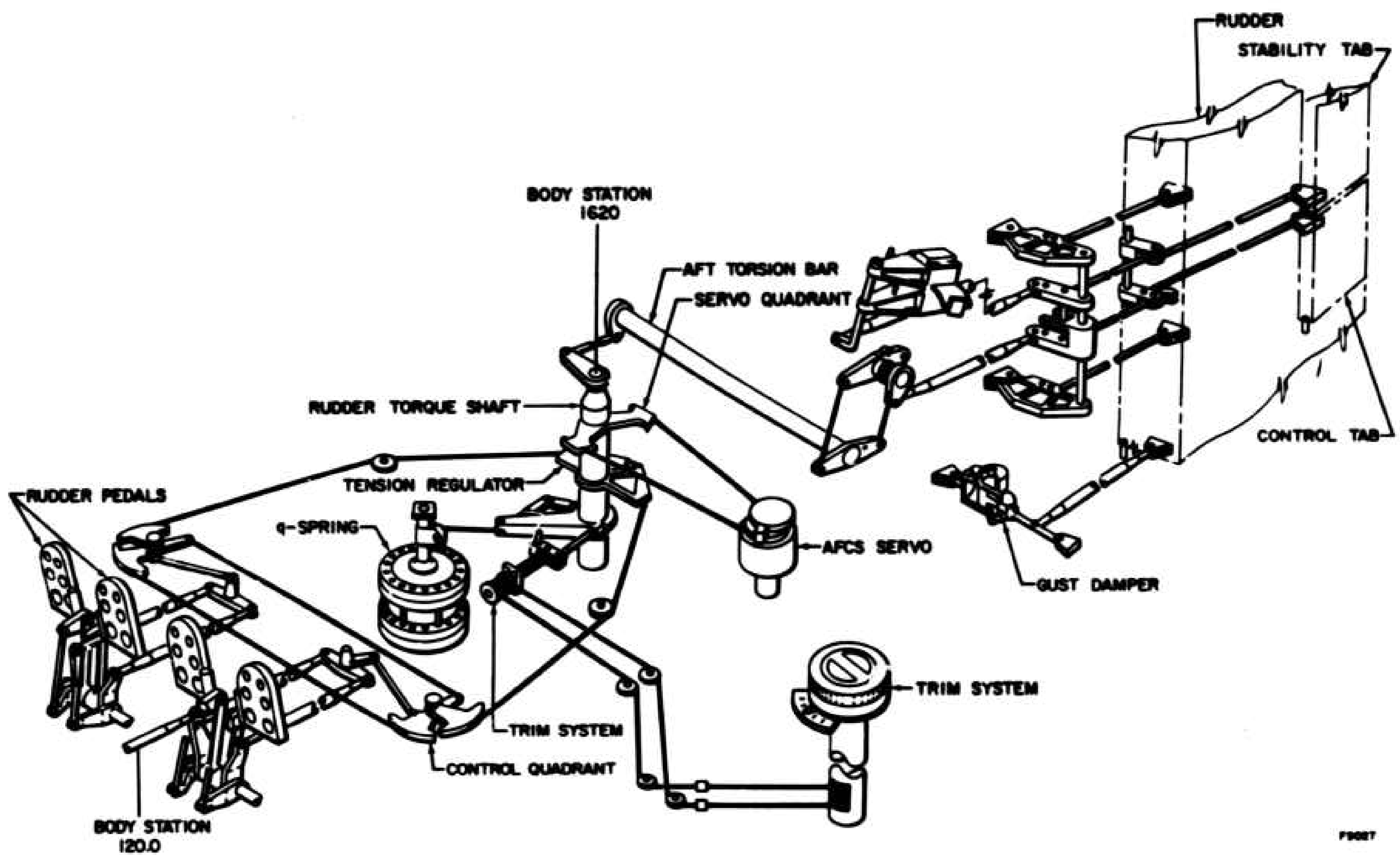


Figure 85
B-52H Rudder Control System (Mechanical)

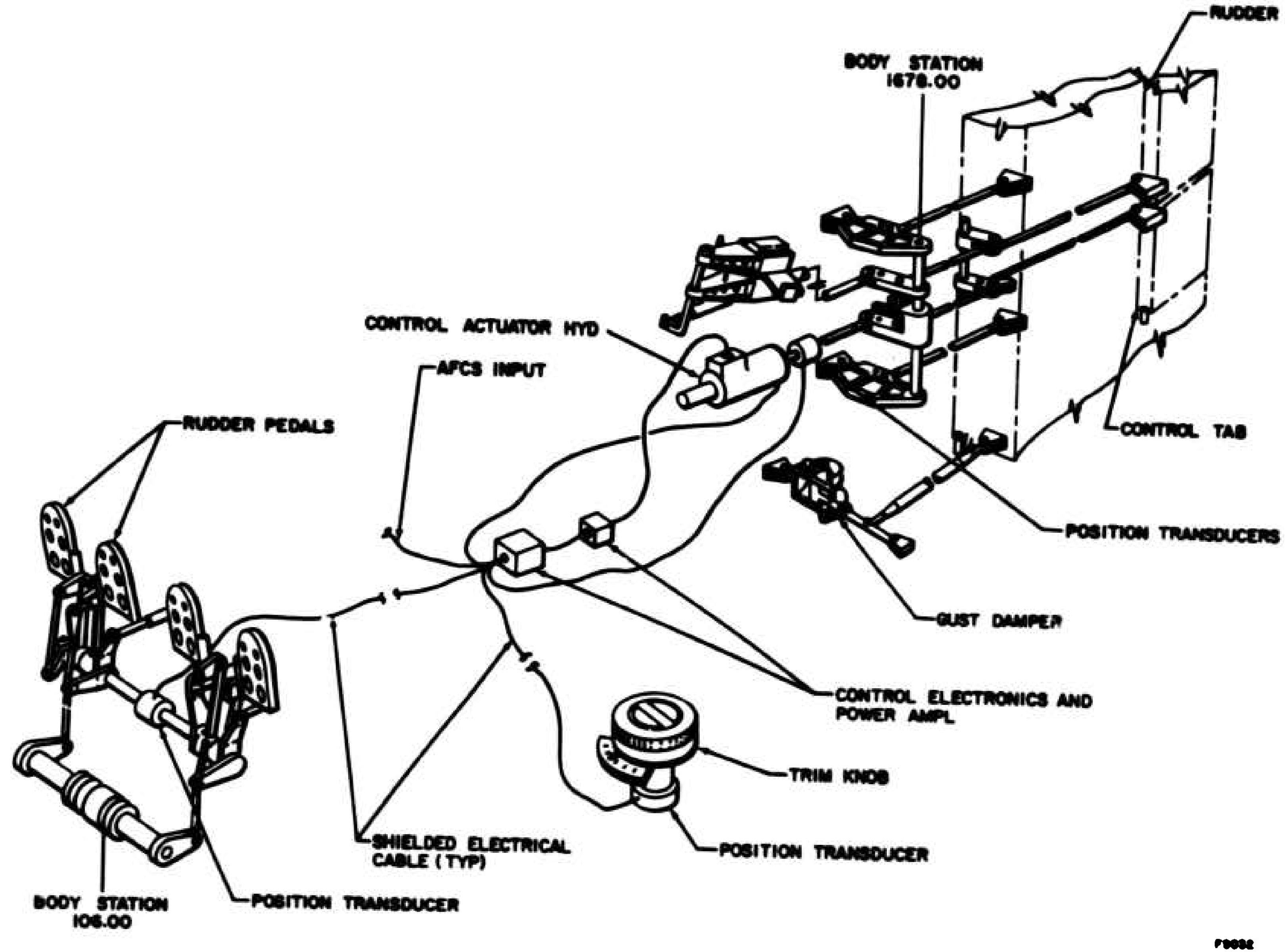


Figure 86
B-52H Rudder Control System (Fly-By-Wire)

Table X presents the weight, volume, and cost of the B-52H mechanical rudder system and the equivalent fly-by-wire system. The fly-by-wire system costs approximately 2.5 times as much as the mechanical system, but it provides a weight savings of 135 pounds and a space savings of 900 cubic inches.

Table XI summarizes the data for comparing the B-52H mechanical control system with the equivalent fly-by-wire system. The fly-by-wire system shows an increase in cost of approximately 100 percent. The large increase reflects the relative simplicity of the mechanical system which employs free-floating tab operated surfaces. For the increase in cost, a fly-by-wire control system provides a reduction of 50 percent in both control system weight and space. The space savings are actually greater because the analysis uses the only displaced volume of the mechanical components and ignores the space allowed for their motion. The fly-by-wire system provides a few hidden and additional benefits. It eliminates the cable rigging maintenance which is required perhaps every 60 flight hours and requires 2 man-days of labor. The question of which system would require more maintenance actions cannot be answered because reliability and maintainability data on this aircraft are not readily available. However, from the Vertol CH-46 analysis described in paragraph VIII.3 of this section, the fly-by-wire system would have a mean time between maintenance actions of 250 hours and have a maintenance time for each flight hour of only 2 minutes. The mission reliability of the two systems is assumed to be equivalent.

2. F-111 FLIGHT CONTROL SYSTEM

Aircraft manual pitch control is achieved by symmetrical motion of the all-movable horizontal tail (elevon). Direct mechanical linkages run from the pilot's control stick to the left and right elevon actuators. Figure 87 shows the direct mechanical linkages and the pitch damper servo which acts as a ground point for mechanical inputs. All pitch stability augmentation and command augmentation signals are input to the pitch damper servo. Trim inputs are either series or parallel. The artificial feel system in the pitch axis consists of fixed springs and electrical signals composed of normal acceleration and angular rate feedback. Figure 87 shows the basic components for the mechanical control system. The components are:

Control stick	Stick position transducer
Upper control quadrant	Manual trim actuator
Lower control quadrant	Series trim actuator
Bellcrank assemblies	Damper servo
Pushrods and end terminals	Elevator surface actuator
Pushrods (adjustable)	Lever assemblies
Feel spring	

A fly-by-wire equivalent to the F-111 mechanical pitch control system is shown in figure 88. The system employs quadruplex position transducers to provide both command signals and actuator feedback signals. Left and right half redundant actuators are driven by quadruplex electronics and servo amplifiers. Stability augmentation, AFCS, series trim and command augmentation signals are summed in the electronics and drive the redundant surface

actuators. Parallel trim is still maintained to provide stick trim displacements if necessary. The basic components for implementing the fly-by-wire system are:

Control stick	Feel spring
Position transducers	Servo amplifier
Control electronics	Surface actuators
Parallel trim	

A comparison of the F-111 mechanical pitch control system and the fly-by-wire equivalent is presented in table XII. The fly-by-wire system increases the cost of the pitch control system by 10 percent over that of the mechanical system. A weight reduction of 98 pounds and space savings of 400 cubic inches result by going to the fly-by-wire system.

The center portion of figure 87 shows the mechanical diagram for the F-111 roll control system. During flight conditions where wing sweep is less than 45 degrees, roll commands are applied to the spoilers and differentially to the horizontal tail. Spoiler authority is 45 degrees. If the wing sweep is greater than 45 degrees, the spoiler system is locked out and all rolling motion is generated from the elevons. Mixing of pitch/roll signals is done in the mechanical mixing mechanism shown in figure 87. Roll stability and command augmentation and roll trim signals are applied to the roll damper servo which also acts as a ground point for pilot mechanical inputs. The manual control of the roll axis of the F-111 consists of the following basic components:

Control stick	Damper servo
Control link	Stick position transducers
Elevon torque shaft	Wing pivot switch
Pushrods and end terminals	Spoiler actuators
Bellcrank assemblies	Spoiler monitor
Pushrods (adjustable)	Spoiler cutoff valve
Low gradient-feel spring	
High gradient-feel spring	

The fly-by-wire equivalent of the F-111 mechanical roll control system can be seen in figure 88. Quadruplex position transducers on the control stick provide manual command signal from the pilot's station to both the spoilers and the horizontal tail actuators. Identical electronics were assumed for the spoilers since the F-111 now has a fly-by-wire spoiler control system. No additional electronics would be required for roll control of the elevon since the pitch channel electronics would be used to drive these actuators.

Table XIII presents a comparison of the F-111 roll control system with an equivalent fly-by-wire system. The fly-by-wire control system for the roll axis of the F-111 can be developed for 30 percent less than the mechanical system. In addition, a weight savings of 75 pounds and a space savings of 450 cubic inches is obtained. This result is obtained because of the compatibility of fly-by-wire control systems, especially when the surfaces they control perform a dual operation.

TABLE X

B-52H RUDDER CONTROL SYSTEM WEIGHT AND CO

Control System Component	Individual			Mechanical Sys		
	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Vo (i
Rudder Pedals	12.45	102	310	2	24.90	20
Control Quadrants	9.28	76.1	231	2	18.56	15
Cable (3/16 in. Diameter)	0.084/ft	0.332/ft	0.505/ft	280 ft	23.52	92
Pushrods (5/8 in. Tube)	0.161/ft	1.32/ft	4.0/ft	46 ft	7.4	60
Turnbuckles	0.279 ea	1.15 ea	7.00 ea	8	2.23	9.
Bellcranks	7.50	61.5	187.0	7 [1]	52.5	43
Torque Shaft	8.74	71.7	218	1	8.74	71
Tension Regulator	14.67	120	365	1	14.67	12
Servo Quadrant	4.65	38.1	116.0	1	4.65	38
Trim Mechanism	5.61	46.0	140	1	5.61	46
Gust Damper	5.8	49	154.1	1	5.8	49
Aft Torsion Bar	5.7	47.0	173.5	1	5.7	47
AFCS Servo	5.80	47.7	1500.0	1	5.80	47
Position Transducer	0.125 ea	1.25 in.^{3}	100.0	---	---	--
Force Feel Spring	11.7	95 in.^{3}	289.0	1	11.7	95
Control Electronics Power Amplifier	1.25	20 in.^{3}	2000	---	---	--
Control Actuator*	7.2	160	5000	---	---	--
Electrical (Shielded) Cable	0.023/ft	0.094/ft	0.10/ft	---	---	--
Cable Pulleys (4 to 6 in.)	0.192	3.14	10.0	8	1.54	25
q-Spring and Equipment	27.5	246.1	748.0	---	---	--
Total System Characteristics	---	---	---	---	193.32	1

*Redundant (Low-Force) Actuator.

[1] Not Common to Both Fly-By-Wi

TABLE X

H RUDDER CONTROL SYSTEM WEIGHT AND COST

	Mechanical System				Fly-By-Wire System			
Cost (dollars)	Number	Weight (lb)	Volume ($in.^3$)	Cost (dollars)	Number	Weight (lb)	Volume ($in.^3$)	Cost (dollars)
310	2	24.90	204	620.0	2	24.90	204	620.0
231	2	18.56	152.2	462.2	---	---	---	---
0.505/ft	280 ft	23.52	92.96	141.40	---	---	---	---
4.0/ft	46 ft	7.4	60.72	184.0	18 ft	2.89	23.76	72.0
7.00 ea	8	2.23	9.2	56.00	2	0.56	2.30	14.00
187.0	7 [1]	52.5	430.5	1309.0	---	---	---	---
218	1	8.74	71.7	218.0	---	---	---	---
365	1	14.67	120	365	---	---	---	---
116.0	1	4.65	38.1	116.0	---	---	---	---
140	1	5.61	46.0	140.0	---	---	---	---
154.1	1	5.8	49	154.1	---	---	---	---
173.5	1	5.7	47.0	173.5	---	---	---	---
1500.0	1	5.80	47.7	1500.0	---	---	---	---
100.0	---	---	---	---	8	1.00	10 $in.^3$	800.0
289.0	1	11.7	95 $in.^3$	289.0	1	11.7	95 $in.^3$	289.0
2000	---	---	---	---	4	5.0	80 $in.^3$	8000
5000	---	---	---	---	1	7.2	160	5000
0.10/ft	---	---	---	---	266.1 ft	6.13	25.03	26.61
10.0	8	1.54	25.12	80.0	---	---	---	---
748.0	---	---	---	---	---	---	---	---
---	---	193.32	1489.2	5808.20	---	59.37	600.1	14,821.6

[1] Not Common to Both Fly-By-Wire and Mechanical System.

TABLE XI

B-52H MECHANICAL/FLY-BY-WIRE CONTROL SYSTEM COMPARISON

Concept / Control Axis	Mechanical System			Fly-By-Wire System		
	Weight (lb)	Volume (ft^3)	Cost (dollars)	Weight (lb)	Volume (ft^3)	Cost (dollars)
Pitch	287.10	1.29	8,054.9	160.66	0.788	17,203.5
Roll	298.1	1.40	17,447.6	143.3	0.710	32,632.5
Yaw	193.3	0.86	5,808.2	59.37	0.347	14,821.6
Total	778.5	3.55	31.310.7	363.33	1.845	64,657.6
Difference	+415.2	+1.71	---	---	---	+33,346.9

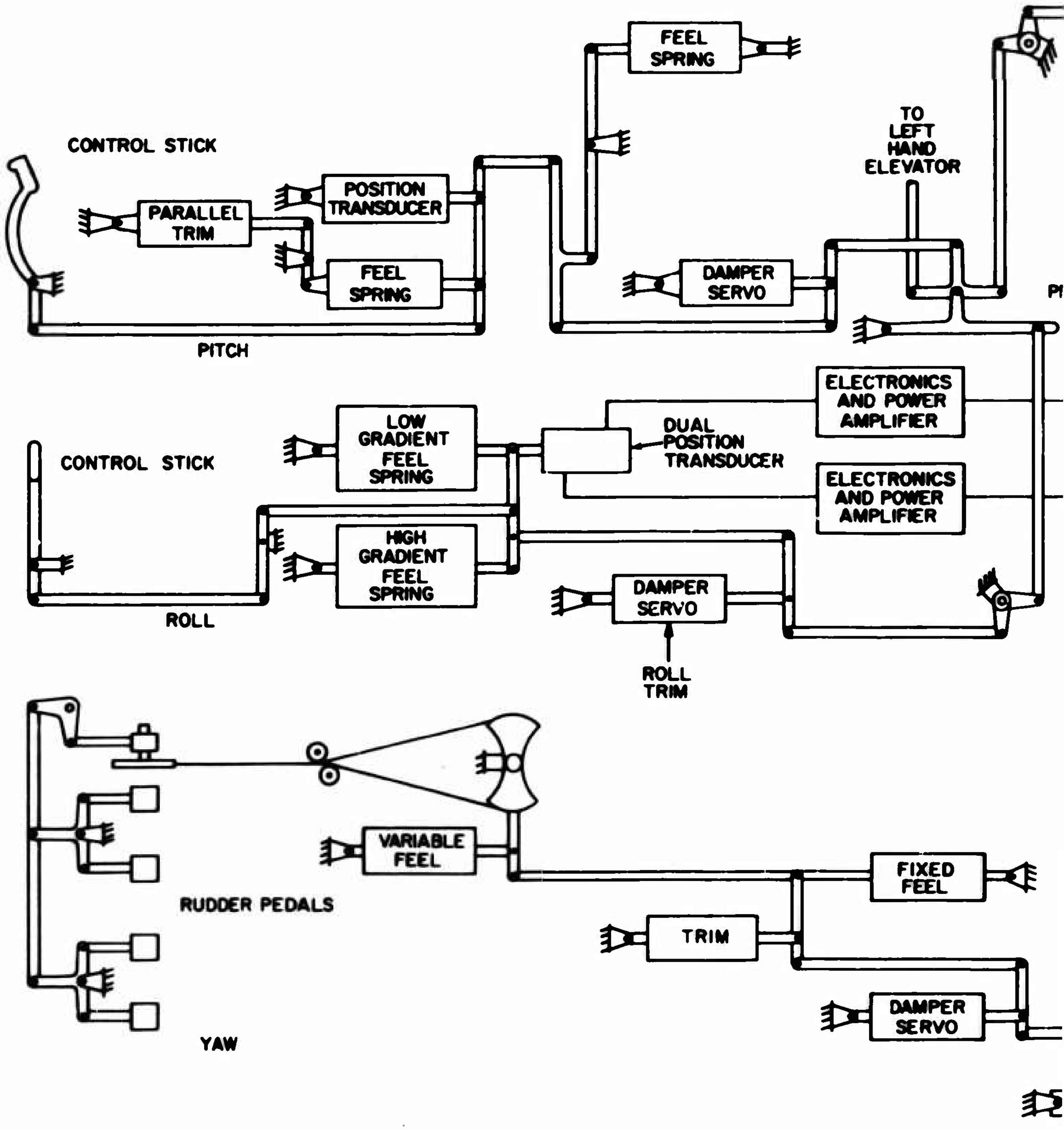
Directional control of the F-111 is achieved by direct mechanical linkages between the rudder pedals and the rudder actuator. Stability augmentation signal and directional trim inputs are series summed with manual inputs. The F-111 rudder system has automatic authority limits which are a function of wing slat position. For slat positions less than 70 percent, the rudder authority is limited to 7.5 degrees to prevent excessive side loads. When the slats extend beyond 70 percent, the feel actuator receives a signal from the slat drive system to change the rudder authority to 30 degrees. This change is made gradually to prevent a noticeable change of rudder pedal position. A monitor circuit warns the pilot when the change is occurring. The mechanical directional control system consists of the following basic components:

Rudder pedals and support	Cable tension regulators
Bellcranks	Variable feel actuators
Quadrants	Fixed gradient feel actuators
Cable	Trim actuator
Turnbuckles	Damper servo
Pulleys and brackets	Surface actuator
Pushrods	Torque tube

A fly-by-wire rudder control system equivalent to the F-111 mechanical control system is shown in figure 88. Pilot input at the rudder pedals generates outputs from quadruplex position transducers to provide control signals to the surface actuator. Electrical feedback signals for the actuator are generated from quadruplex position transducers mounted on the actuator output. All electronics and servo actuators are quadruplex. Table XIV presents a comparison of the F-111 directional control system with a fly-by-wire system. The fly-by-wire system and the mechanical system cost approximately the same, but the fly-by-wire system provides a weight savings of over 100 pounds and a space savings of over 600 cubic inches.

A comparison of the full primary flight control system for the F-111 with an equivalent fly-by-wire system with respect to cost, weight and space is presented in table XV. The fly-by-wire system resulted in a weight savings of 277.3 pounds and a space savings of 0.85 cubic feet. Cost figures for the two systems were nearly equal with the fly-by-wire system being approximately 4 percent less. General Dynamics, during an intensive weight saving program, made a study on the F-111 aircraft similar to the one described herein. Their weight, shown in parentheses, corroborate our results.

A comparison of tables XI and XV for the B-52H and the F-111 respectively, shows that fly-by-wire systems in general would be smaller and lighter than their mechanical equivalent. The trends in cost of the systems show that mechanical systems are cheaper for simple systems and fly-by-wire systems are cheaper as control systems become more complex. A discussion with the B-70 controls group at North American further corroborated this cost trend. The fly-by-wire control system for a B-70 would reduce control system weight by approximately 675 pounds and would result in a 90 percent saving in design manhours.



CONTROL STICK
PARALLEL TRIM
POSITION TRANSDUCER
FEEL SPRING
FEEL SPRING
TO LEFT HAND ELEVATOR
DAMPER SERVO
PITCH
CONTROL STICK
LOW GRADIENT FEEL SPRING
HIGH GRADIENT FEEL SPRING
DUAL POSITION TRANSDUCER
ELECTRONICS AND POWER AMPLIFIER
ELECTRONICS AND POWER AMPLIFIER
DAMPER SERVO
ROLL TRIM
ROLL
VARIABLE FEEL
RUDDER PEDALS
FIXED FEEL
TRIM
DAMPER SERVO
YAW

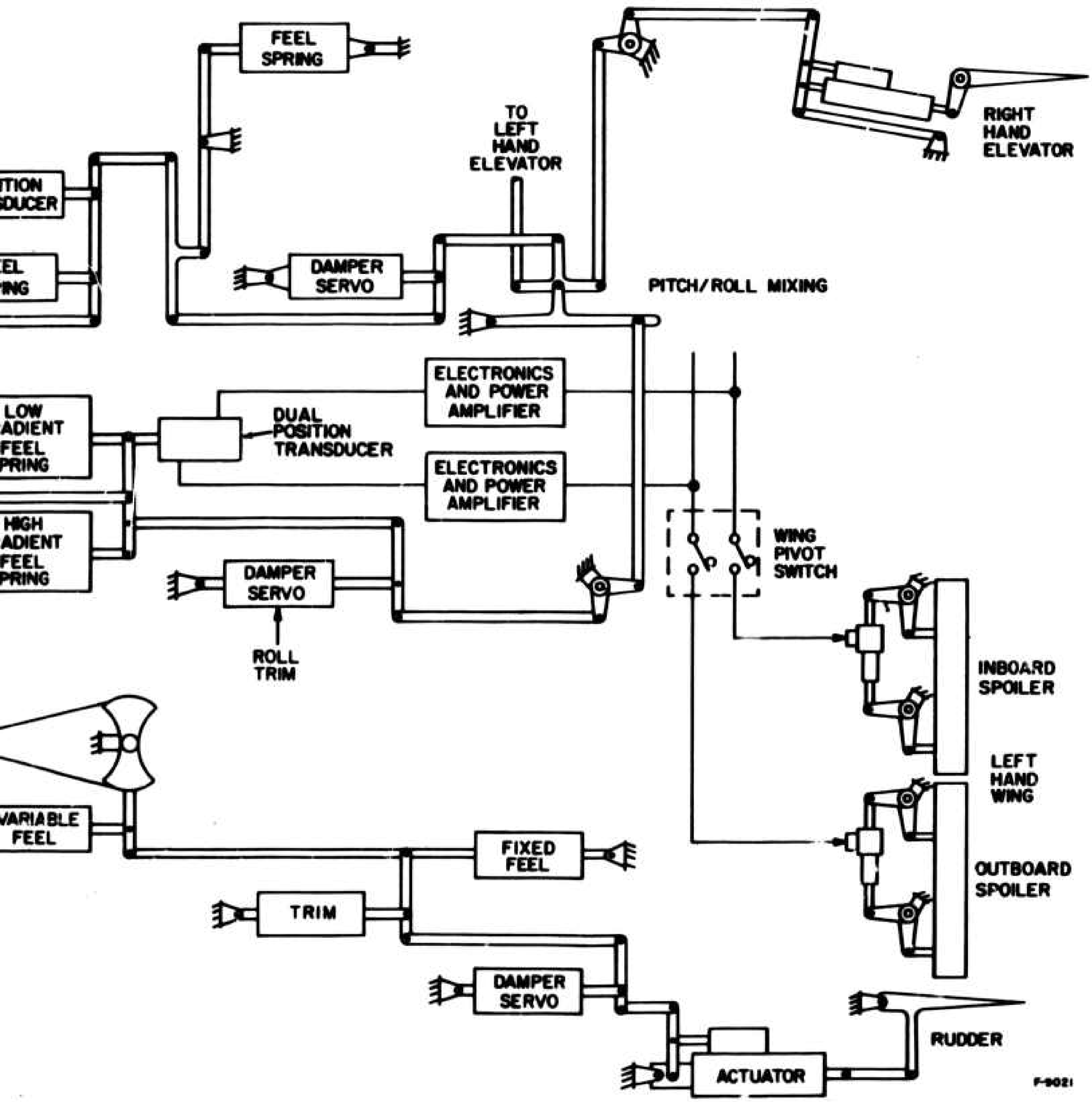
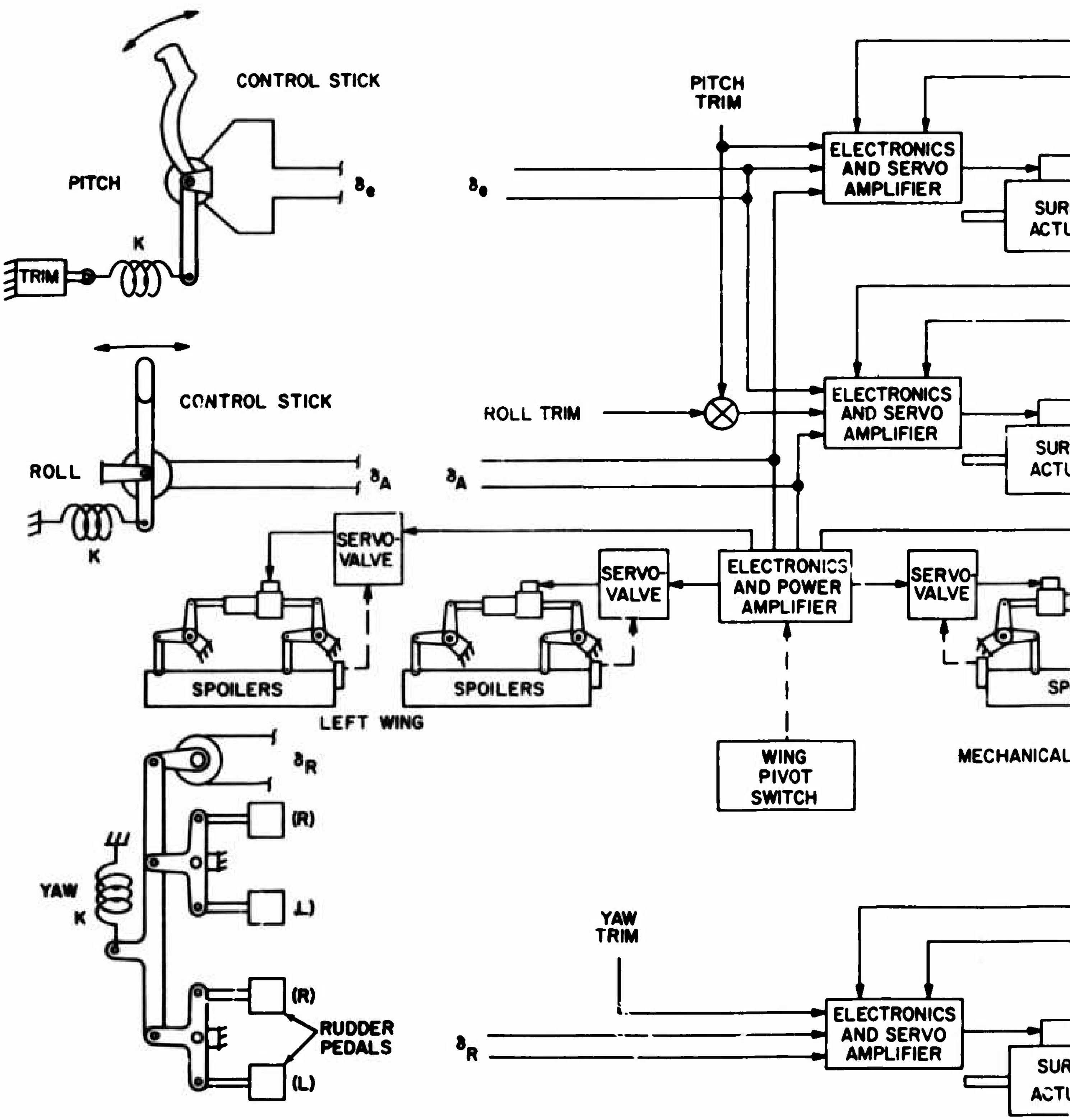


Figure 87
F-111 A/B Flight Control
Schematic (Mechanical)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ELECTRONICS AND POWER AMPLIFIER
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ELECTRONICS AND SERVO AMPLIFIER
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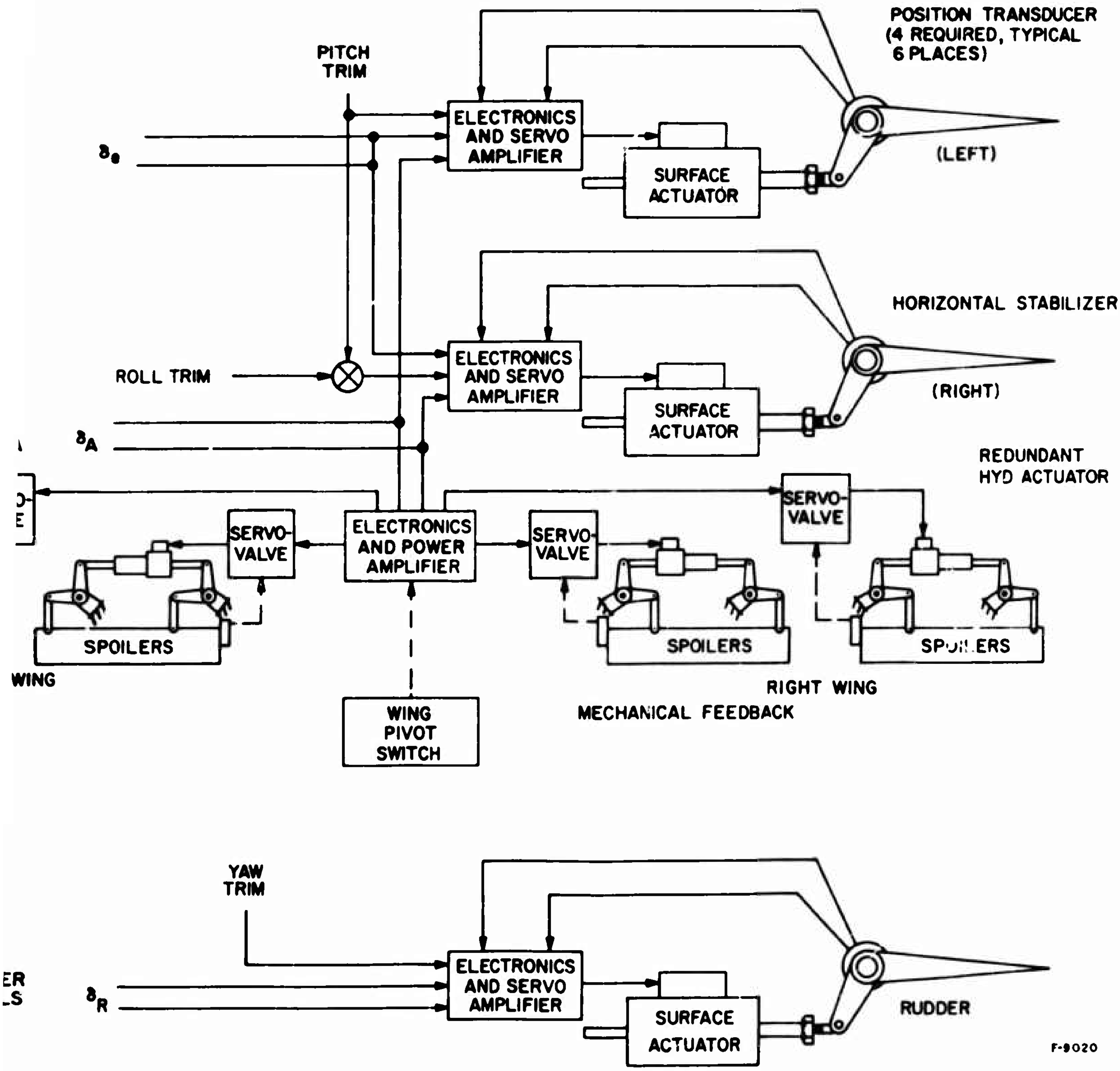

Figure 88
F-111 A/B Flight Control Schematic
(Fly-By-Wire)

TABLE XII

F-111 PITCH CONTROL SYSTEM WEIGHT AND

Control System Component	Individual			Mechanical S	
	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)
Control Stick	3.41	28.0	421.8	2	6.82
Upper Control Quadrant	7.61	62.9	273.6	1	7.61
Lower Control Quadrant	4.80	39.3	170.9	1	4.80
Fixed Length Pushrod Assembly	0.65/ft	4.18/ft	8.4/ft	76.5 ft	49.73
Variable Length Pushrod Assembly	0.73/ft	5.32/ft	13.14/ft	12.3 ft	8.97
Bellcrank (2 in. Arm)	0.39	2.70	14.10	4	1.56
Bellcrank (3 in. Arm)	0.48	3.26	16.98	6	2.88
Bellcrank (3-1/2 in. Arm)	0.52	3.54	18.44	7	3.64
Bellcrank (4 in. Arm)	0.56	3.82	19.90	20	11.20
Bellcrank (5 in. Arm)	0.64	4.38	22.82	4	2.56
Bellcrank (6 in. Arm)	0.72	4.94	25.74	4	2.88
Artificial Feel Spring	7.31	60.0	261.0	1	7.31
Position Transducer	0.125	1.25	100.0	3	0.375
Parallel Trim Actuator	6.8	30.0	1800.0	1	6.8
Series Trim Actuator	6.8	300.0	1800.0	1	6.8
Damper* Servo Actuator	18.35	64.0	6000.0	1	18.35
Stabilizer Surface Actuator	9.68	68.30	4500.0	2	19.37

*Redundant Hydraulic Actuator (Low Force)

TABLE XII

L1 PITCH CONTROL SYSTEM WEIGHT AND COST

	Mechanical System				Fly-By-Wire System			
Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)
421.8	2	6.82	56	843.6	2	6.82	56.0	843.6
273.6	1	7.61	62.9	273.6	1	7.61	62.9	273.6
170.9	1	4.80	39.3	170.9	---	---	---	---
8.4/ft	76.5 ft	49.73	319.8	642.6	---	---	---	---
13.14/ft	12.3 ft	8.97	65.4	161.6	---	---	---	---
14.10	4	1.56	10.8	56.4	---	---	---	---
16.98	6	2.88	19.6	101.9	4	1.95	13.05	67.91
18.44	7	3.64	24.7	129.1	---	---	---	---
19.90	20	11.20	76.4	398.6	---	---	---	---
22.82	4	2.56	17.5	91.31	---	---	---	---
25.74	4	2.88	19.76	102.96	---	---	---	---
261.0	1	7.31	60.0	261.0	1	7.31	60.0	261.0
100.0	3	0.375	3.75	300.0	12	1.5	15.0	1200.0
1800.0	1	6.8	30.0	1800.0	1	6.8	30.0	1800.0
1800.0	1	6.8	30.0	1800.0	---	---	---	---
6000.0	1	18.35	64.0	6000.0	---	---	---	---
4500.0	2	19.37	136.6	9000.0	---	---	---	---

TABLE XII (cont)

F-111 PITCH CONTROL SYSTEM WEIGHT AND (

Control System Component	Individual			Mechanical Sy	
	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)
Lever Arm Assembly	1.77	14.5	63.1	1	1.77
Control Electronics/ Channel Servo Power Amplifier/Channel	1.250	20.0	2000.0	---	---
Redundant Stabilizer Actuator	12.3	160.0	6000.0	---	---
Electronic Transmission Cable (Shielded #22)	0.023/ft	0.094/ft	0.10/ft	---	---
Total System Characteristics	---	---	---	---	163.24

TABLE XII (cont)

PITCH CONTROL SYSTEM WEIGHT AND COST

	Mechanical System				Fly-By-Wire System			
Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)
63.1	1	1.77	14.5	63.1	---	---	---	---
2000.0	---	---	---	---	4	5.5	90.0	10,000.0
6000.0	---	---	---	---	2	24.6	320.0	12,000.0
0.10/ft	---	---	---	---	187.0	4.3	17.6	18.7
---	- -	163.24	1071.0	22,196.7	---	66.4	654.6	26,464.8

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TABLE XIII

F-111 ROLL CONTROL SYSTEM WEIGHT AND C

Control System Component	Individual			Mechanical Sy		
	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	V (
Control Stick	3.41	28.0	421.8	2	6.82	5
Fwd Control Link	4.09	33.5	146.1	1	4.09	3
Roll Torque Shaft	3.69	30.3	132.1	1	3.69	3
Fixed Length Pushrod Assembly (1.0 in. O.D. x 1/8 Tube)	0.65/ft	4.18/ft	8.4/ft	34.9 ft	22.69	1
Variable Length Pushrod Assembly (1.0 in. O.D. x 1/8 in. Tube)	0.73/ft	5.32/ft	13.14/ft	18.1 ft	13.21	9
Position Transducer	0.125	1.25	100.0	3	0.375	[illegible]
Lever Arm Links	3.41	28.1	122.1	1	3.41	2
Spoiler Surface Actuator	7.1	58.0	3100.0	4	28.4	2
Damper* Servo Actuator	18.35	64.0	6000.0	1	18.35	6
Bellcrank (2 in. Arm)	0.39	2.7	14.10	3	1.17	8
Bellcrank (3 in. Arm)	0.48	3.26	16.98	6	2.88	1
Bellcrank (3-1/2 in. Arm)	0.52	3.54	18.44	1	0.52	[illegible]
Bellcrank (4 in. Arm)	0.56	3.82	19.90	7	3.92	[illegible]
Bellcrank (5 in. Arm)	0.64	4.38	22.80	4	2.56	[illegible]
Bellcrank (6 in. Arm)	0.72	4.94	25.74	1	0.72	[illegible]
Control Electronics/Channel	1.0	6.0	1600.0	2	2.0	

*Redundant Hydraulic Actuator (Low Force)

TABLE XIII

11 ROLL CONTROL SYSTEM WEIGHT AND COST

	Mechanical System				Fly-By-Wire System			
Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)
421.8	2	6.82	56.0	843.6	2	6.82	56.0	843.6
146.1	1	4.09	33.5	146.1	1	4.09	33.5	146.1
132.1	1	3.69	30.3	132.1	1	3.69	30.3	132.1
8.4/ft	34.9 ft	22.69	145.8	331.1	---	---	---	---
13.14/ft	18.1 ft	13.21	96.3	237.5	---	---	---	---
100.0	3	0.375	3.75	300.0	4	0.5	5.0	400.0
122.1	1	3.41	28.1	122.1	---	---	---	---
3100.0	4	28.4	232.0	12,400.	4	28.4	232.0	12,400
6000.0	1	18.35	64.0	6000.0	---	---	---	---
14.10	3	1.17	8.10	42.3	---	---	---	---
16.98	6	2.88	19.56	101.8	---	---	---	---
18.44	1	0.52	3.54	18.44	---	---	---	---
19.90	7	3.92	26.7	139.3	4	2.24	15.3	80.21
22.80	4	2.56	17.52	91.20	---	---	---	---
25.74	1	0.72	4.94	25.74	---	---	---	---
1600.0	2	2.0	12.0	3200.0	2	2.0	12.0	3200.0

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TABLE XIII (cont)

F-111 ROLL CONTROL SYSTEM WEIGHT AND

Control System Component	Individual			Mechanical S	
	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)
Servo Power Amplifier/Channel	0.5	6.0	500.0	2	1.0
High Gradient Artificial Feel Spring	11.6	72.0	374.1	1	11.6
Low Gradient Artificial Feel Spring	7.31	60.0	261.0	1	7.31
Electronics Transmission Shielded Cable (#22)	0.023/ft	0.094/ft	0.10/ft	87 ft	2.01
Total System Characteristics	---	---	---	---	136.20

TABLE XIII (cont)

11 ROLL CONTROL SYSTEM WEIGHT AND COST

	Mechanical System				Fly-By-Wire			
Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)
500.0	2	1.0	12.0	1000.0	2	1.0	12.0	2000.0
374.1	1	11.6	72.0	374.1	---	---	---	---
261.0	1	7.31	60.0	261.0	1	7.31	60.0	261.0
0.10/ft	87 ft	2.01	8.19	8.70	227 ft	5.22	21.34	22.7
---	---	136.20	933.25	25,775.1	---	61.27	477.44	19,485.7

TABLE XIV

F-111 YAW CONTROL SYSTEM WEIGHT AND

Control System Component	Individual			Mechanical S	
	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)
Rudder Pedal System	12.45	102.1	310.0	2	24.9
Rudder Pedal Supports	3.90	32.0	139.0	2	7.80
Fwd, Rudder Bellcrank	2.45	18.3	95.6	2	4.90
Forward Cable Quadrant	3.21	26.1	97.1	1	3.21
Control Cable (3/16 in. Diameter Stl)	0.084/ft	0.33/ft	0.505/ft	78 ft	6.55
Turnbuckles	0.279	1.15	7.0	6	1.67
Cable Pulleys (4 to 6 in. Diameter)	0.192	3.14	10.0	16	3.07
Tension Regulator Quadrant	9.3	87.0	265.0	1	9.3
Variable Feel Actuator	4.94	27.9	876.0	1	4.94
Fixed Gradient Force Feel Spring	6.31	87.0	261.0	1	6.31
Series Trim Actuator	6.8	30.0	1800.0	1	6.8
Damper Servo Actuator	18.35	64.0	6000.0	1	18.35
Rudder Surface Actuator	9.68	68.3	4500.0	1	9.68
Fixed Length Pushrod Assembly	0.65/ft	4.18/ft	8.4/ft	41.7 ft	27.11
Variable Length Pushrod Assembly	0.73/ft	5.32/ft	13.14/ft	12.04	8.79
Bellcrank (2 in. Arm)	0.39	2.70	14.1	3	1.17
Bellcrank (3 in. Arm)	0.48	3.26	16.98	3	1.44

TABLE XIV

'-111 YAW CONTROL SYSTEM WEIGHT AND COST

	Mechanical System				Fly-By-Wire System			
Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)
310.0	2	24.9	204.2	620.0	2	24.9	204.2	620.0
139.0	2	7.80	64.0	278.0	2	7.80	64.0	278.0
95.6	2	4.90	36.6	191.2	2	4.9	36.6	191.2
97.1	1	3.21	26.1	97.1	---	---	---	---
0.505/ft	78 ft	6.55	25.9	39.4	---	---	---	---
7.0	6	1.67	6.9	42.0	---	---	---	---
10.0	16	3.07	50.2	160.0	---	---	---	---
265.0	1	9.3	87.0	265.0	---	---	---	---
876.0	1	4.94	27.9	876.0	---	---	- -	---
261.0	1	6.31	87.0	261.0	1	6.31	87.0	261.0
1800.0	1	6.8	30.0	1800.0	---	---	---	---
6000.0	1	18.35	64.0	6000.0	---	---	---	---
4500.0	1	9.68	68.3	4500.0	---	---	---	---
8.4/ft	41.7 ft	27.11	200.2	350.4	---	---	---	---
13.14/ft	12.04	8.79	69.9	158.2	---	---	---	---
14.1	3	1.17	8.10	42.30	---	---	---	---
16.98	3	1.44	9.78	50.94	---	---	---	---

TABLE XIV (cont)

F-111 YAW CONTROL SYSTEM WEIGHT AND COS

Control System Component	Individual			Mechanical Sys		
	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Vo (i
Bellcrank (3-1/2 in. Arm)	0.52	3.54	18.44	7	3.64	24
Bellcrank (4 in. Arm)	0.56	3.82	19.90	6	3.36	22
Bellcrank (3 in. Arm)	0.64	4.38	22.80	13	8.32	56
Bellcrank (6 in. Arm)	0.72	4.94	25.74	2	1.44	9.
Bellcrank (8 in. Arm)	0.82	6.09	31.71	5	4.10	30
Aft Torsion Bar	4.32	35.7	232.1	1	4.32	35
Control Links (Aft)	1.44	9.88	81.3	2	2.88	19
Position Transducer	0.125	1.25	100.0	---	---	--
Control Electronics/ Channel	1.0	6.0	1600.0	---	---	--
Servo Power Ampl/Channel	0.5	6.0	500.0	---	---	--
Redundant Rudder Actuator	12.3	160.0	6000.0	---	---	--
Electronics Transmission Cable (Shielded #22)	0.023/ft	0.094/ft	0.10/ft	---	---	--
Total System Characteristics	---	---	---	---	174.1	12

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TABLE XIV (cont)

11 YAW CONTROL SYSTEM WEIGHT AND COST

	Mechanical System				Fly-By-Wire System			
Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)	Number	Weight (lb)	Volume (in.^{3})	Cost (dollars)
18.44	7	3.64	24.78	129.08	4	2.08	14.15	73.72
19.90	6	3.36	22.92	119.40	---	---	---	---
22.80	13	8.32	56.94	296.70	---	---	---	---
25.74	2	1.44	9.88	51.48	---	---	---	---
31.71	5	4.10	30.45	158.55	---	---	---	---
232.1	1	4.32	35.7	232.1	---	---	---	---
81.3	2	2.88	19.76	162.6	---	---	---	---
100.0	---	---	---	---	8	1.0	10.0	800.0
1600.0	---	---	---	---	4	5.0	80.0	8000.0
500.0	---	---	---	---	---	---	---	---
6000.0	---	---	---	---	1	12.3	160.0	6000.0
0.10/ft	---	---	---	---	187.0	4.3	17.6	18.70
---	---	174.1	1266.5	16,881.5	---	68.6	663.6	16,242.6

TABLE XV

F-111 MECHANICAL/FLY-BY-WIRE CONTROL SYSTEM COMPARISON

Control Axis	Concept	Mechanical System			Fly-By-Wire System		
		Weight (lb)	Volume (ft^3)	Cost (dollars)	Weight (lb)	Volume (ft^3)	Cost (dollars)
Pitch	---	163.24	0.620	22,196.7	66.4	0.385	26,464.8
Roll	---	136.2	0.540	25,775.1	61.27	0.276	19,485.7
Yaw	---	174.1	0.733	16,881.5	68.6	0.384	16,242.6
Total	---	(367)* 473.54	1.89	64,853.3	(200)* 196.27	1.045	62,193.1
Difference	---	+277.3	+0.85	+2,659.9	---	---	---

*Parenthetical weights are the results of a similar analysis made by General Dynamics. The lower weight of the mechanical system is the result of an intensive weight saving program.

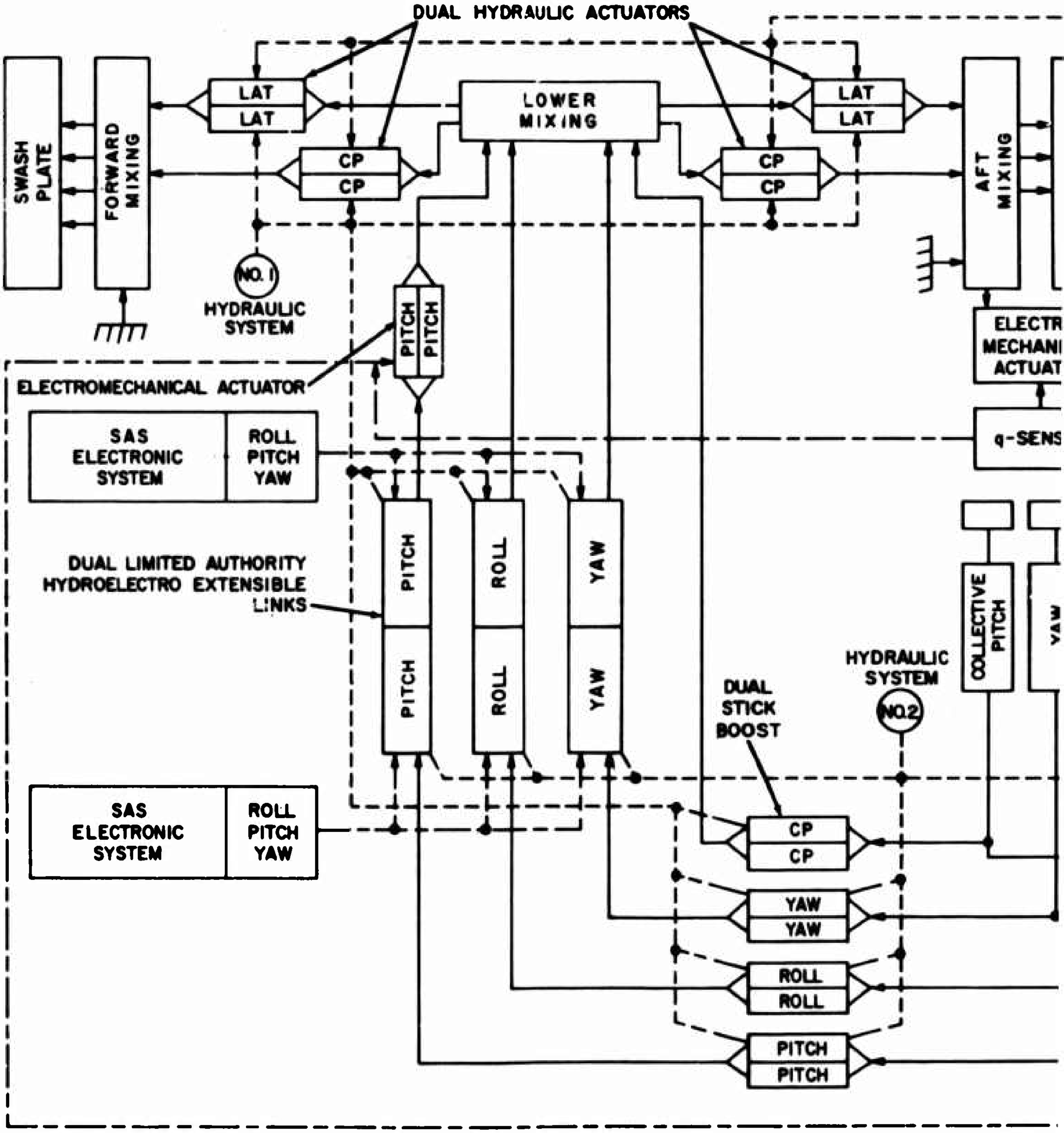
If the trends resulting from the comparison of mechanical and fly-by-wire systems are extrapolated to the advanced aircraft of the 1970's, one can easily see that fly-by-wire control systems will be smaller, lighter, and cheaper.

3. CH-46 FLIGHT CONTROL SYSTEM

The existing cockpit controls of the CH-46 are the cyclic control stick, the directional pedals, and the collective stick. Fore and aft motion of the cyclic stick controls forward velocity or pitch angle, lateral motion of the cyclic stick controls lateral velocity or roll, the directional pedals control yaw rate or sideslip, and the collective stick controls altitude. The primary flight controls consist of the lateral cyclic and collective pitch of the two rotors. Differential collective pitch controls the pitching moment by increasing (decreasing) the lift force of the forward rotor and decreasing (increasing) the lift of the aft rotor. Equal changes in lateral cyclic pitch of the rotors controls the rolling moment. Differential lateral cyclic pitch controls the yaw moment. The total lift force is controlled either by equal change in the collective pitch in both rotors or by tilting rotor disc angle of attack during forward flight. Forces in the horizontal plane are produced by tilting the lift vector through pitch and roll commands. Figure 89 shows the CH-46A flight control system block diagram.

The fly-by-wire system replaces the current mechanical control system except for the cockpit controls and the feel system. This includes the following existing equipment: the control stick dual boost actuators (4), the series stability augmentation actuators (6), pitch series trim actuator (1), the mixing unit, the swash plate dual boost actuators (4), and all of the related interconnecting links. The fly-by-wire system replaces all of this equipment with stick position transducers (4), provisions for control signal shaping, mixing electronics, and swash plate actuators (4). The swash plate actuators combine all of the functions of the existing swash plate actuators, SAS actuators, and series trim actuators, and thus eliminate a considerable amount of complexity and weight. Eliminating the mechanical control linkages eliminates the need for the stick boost actuators. The electronics sum, shape, mix, and blend the stick position, SAS, and ASE (automatic stabilization equipment) signals to generate the proper control signals for the actuators. Various nonlinear or variable control functions can be inserted in the shaping electronics to evaluate their effects on control response or to tailor the response to the pilot's taste. The ASE signals interface mechanically with the cockpit controls through electromechanical servos to provide parallel stick motions. As an alternate configuration, the ASE could be interfaced electrically with the fly-by-wire system (as shown in figure 90), and thus further simplify the control system by eliminating the trim servos. Since trim motion feedback to the stick would also be eliminated by this approach, it is not recommended for this application. A simplified block diagram is shown in figure 90; implementation of quadruplex redundant electronics in the longitudinal axis is shown in figure 91. (The lateral axis would be similar.)

The system components can be grouped into control stick transducers, electronics, and actuators. The design employs quadruplex redundancy. Each of the four control stick or pedal transducers consist of a quadruplex linear


DUAL HYDRAULIC ACTUATORS
SWASH PLATE
FORWARD MIXING
LAT
LAT
CP
CP
LOWER MIXING
LAT
LAT
CP
CP
AFT MIXING
NO. 1
HYDRAULIC SYSTEM
PITCH
PITCH
ELECTROMECHANICAL ACTUATOR
SAS ELECTRONIC SYSTEM
ROLL PITCH YAW
DUAL LIMITED AUTHORITY HYDROELECTRO EXTENSIBLE LINKS
PITCH
ROLL
YAW
PITCH
ROLL
YAW
COLLECTIVE PITCH
HYDRAULIC SYSTEM
NO.2
DUAL STICK BOOST
SAS ELECTRONIC SYSTEM
ROLL PITCH YAW
CP
CP
YAW
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PITCH

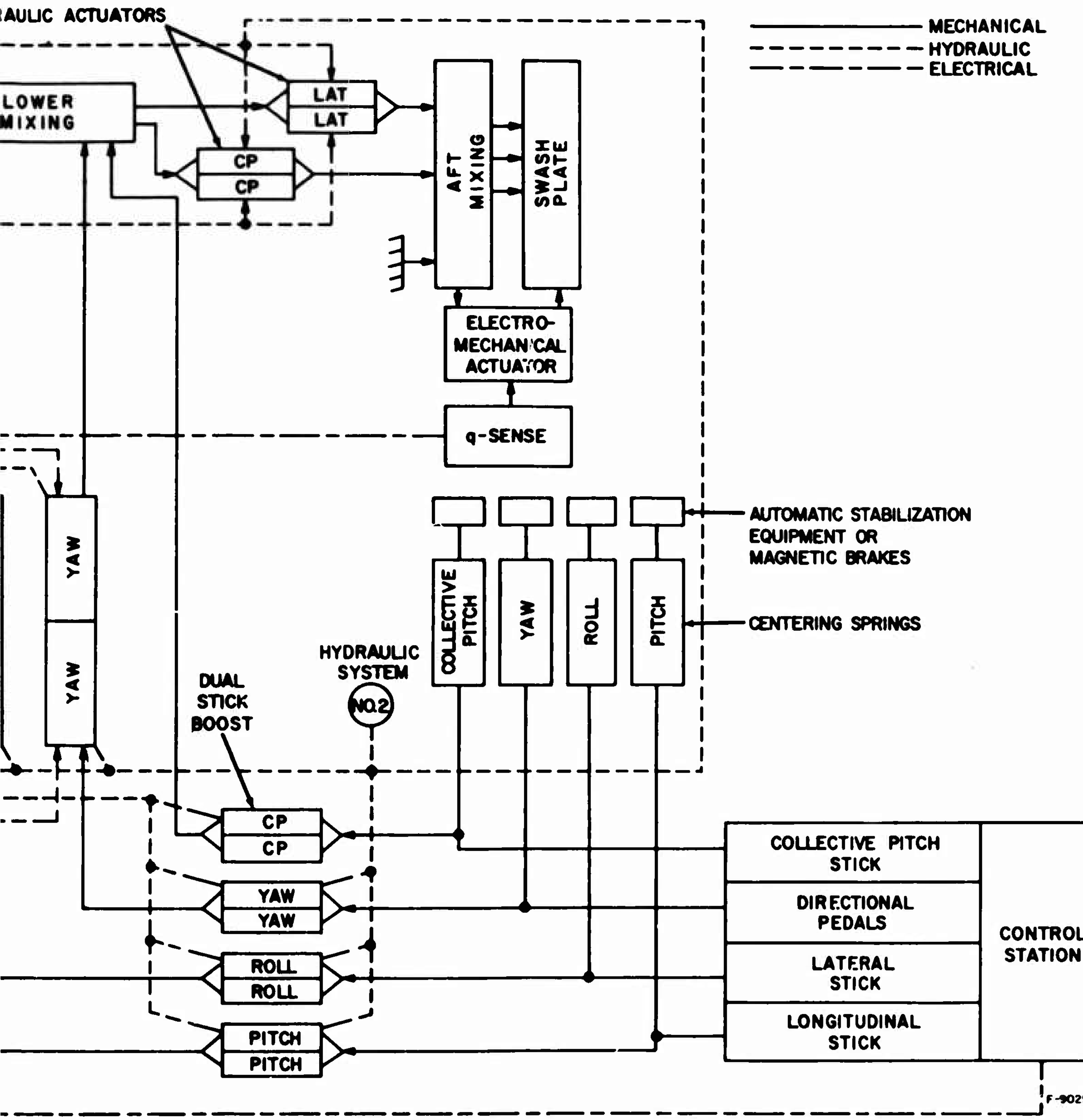


Figure 89
CH-46A Flight Control System
Block Diagram

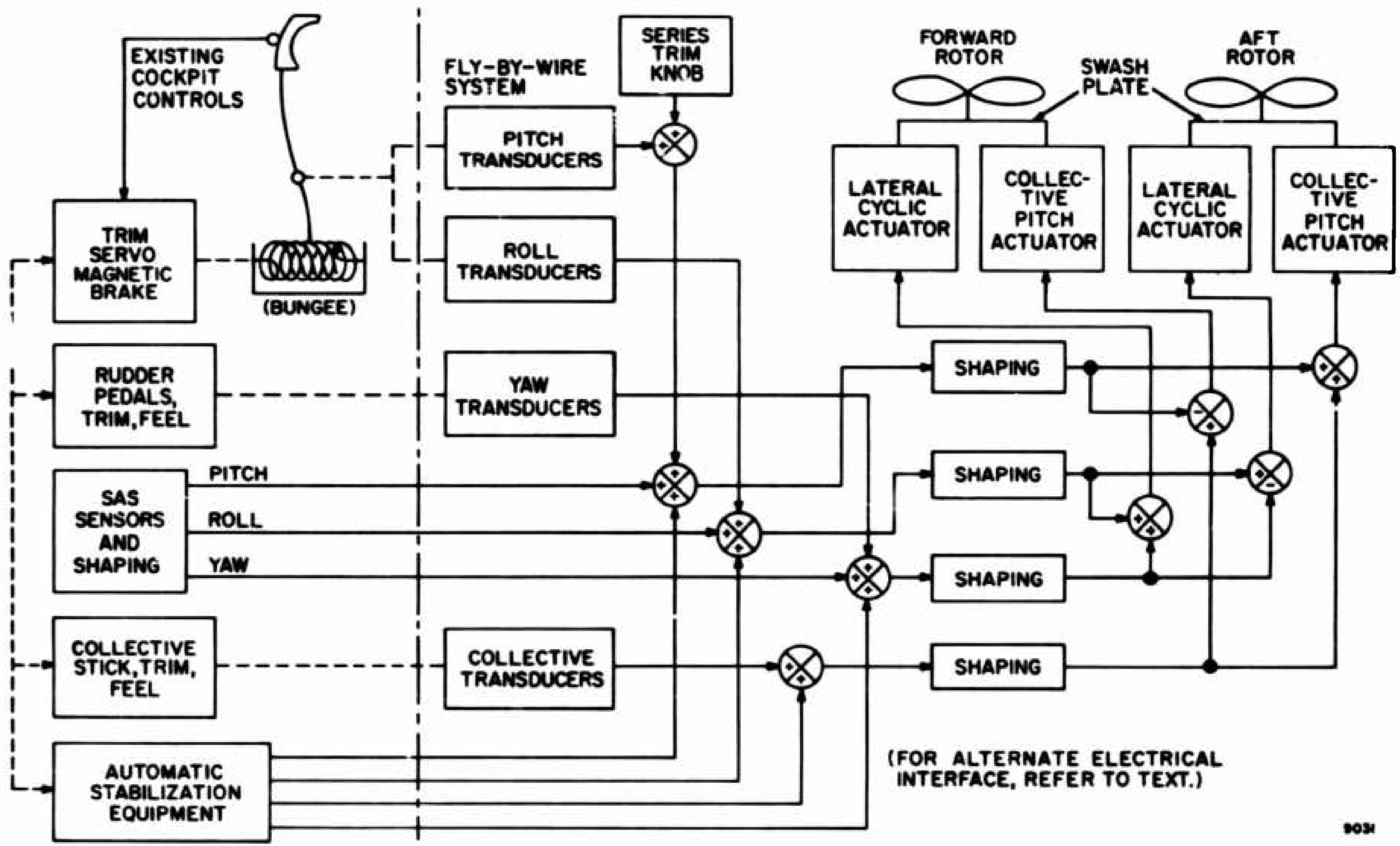


Figure 90
CH-46 Fly-By-Wire System - Simplified Block Diagram

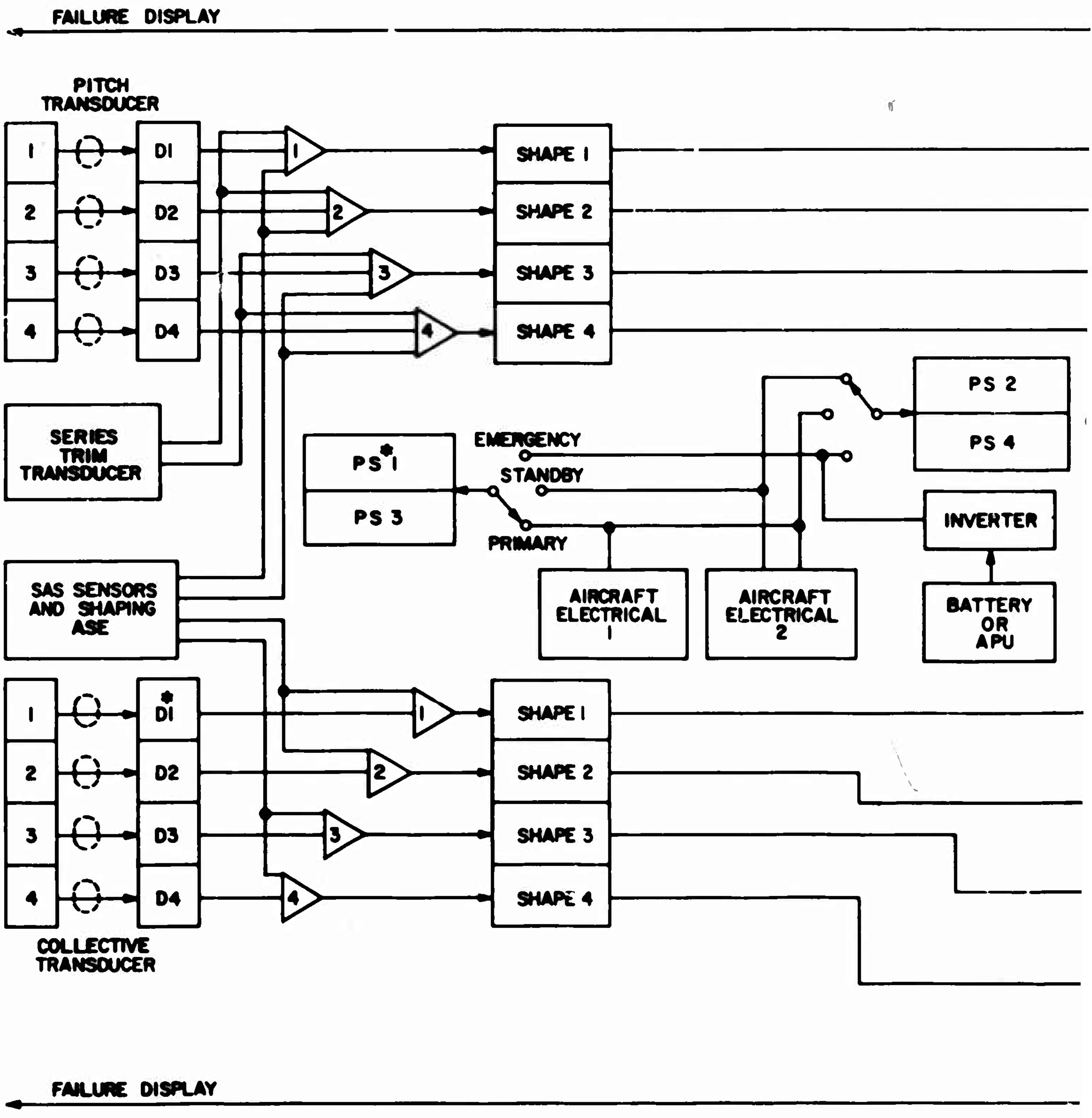
synchro or LVDT depending on the space available and motion required. Each transducer occupies about 3 cubic inches and weighs 6 ounces. DC electronics are used as discussed in Sections V and VI. The electronics are packaged by channels; that is, all channel A's for the four axes are packaged in one chassis, channel B's in another, and so on (a total of four units). Each unit measures approximately 6 inches x 4 inches x 4-1/2 inches and weighs approximately 5 pounds for a total volume of 108 cubic inches and weight of 20 pounds. (An alternate approach which could be used packages the channels by pairs resulting in two units each of which measures approximately 12 inches x 4 inches x 4-1/2 inches and weighs 9 pounds.) The total electrical cable weight is estimated at 35 pounds.

The system employs the quadruplex fail-passive actuator which is described in paragraph VI.5, model 4A. Each of the four servos measures approximately 20 inches x 3 inches x 6 inches and weighs approximately 15 pounds. These estimates assume the use of 1500 psi hydraulic supplies. The size would be smaller, of course, if 3000 psi supplies were used. The total system volume and weight then becomes 1950 cubic inches (1.13 cubic feet) and 115 pounds. The weight of the CH-46 mechanical system is 500 pounds, and the slightly larger CH-47 system (which the above system could also replace) weighs 833 pounds. This represents a savings of 385 and 718 pounds, respectively.

The summary tables of predicted failure rates of the system components are listed in tables XVI and XVII. These values were obtained by using the component failure rates in MIL-HDBK-217A combined with estimates average stress levels and preliminary parts count. In cases where MIL-HDBK-217A does not contain data on a part, Avco Reliability Engineering Data Series, or Sperry field data is used. The estimated component or subassembly failure rates used in this proposal are summarized in table XVI.

Because of control signal mixing, the reliability of one longitudinal channel, one thrust channel, one forward collective inner servo, and one aft collective inner servo are in series, as is the reliability of one lateral channel, one directional channel, one forward cyclic inner servo and one aft cyclic inner servo. That is, the failures in these channels are detected by the same monitor at the actuator. Since the monitor cannot differentiate between them, a failure in any part of the chain is construed as a failure of the complete chain. Call the total failure rate of this signal channel λ_s. Assuming that two channels of the four must work for a successful mission, the probability of failure $P_f = 4\ (\lambda_s t)^3$ for both the pitch-collective and roll-yaw axes. The boost actuator calculation, which is separate, assumes that the actuator cannot jam and that one actuator of the two must work for success; therefore, P_f (actuator) $= (\lambda_a t)^2$ for each tandem actuator. For the complete system then, the probability of failure for a 1-hour mission is

$$P_f = 2 \times 4(\lambda_s t)^3 + 4(\lambda_a t)^2$$


FAILURE DISPLAY
PITCH TRANSDUCER
1
2
3
4
D1
D2
D3
D4
1
2
3
4
SHAPE 1
SHAPE 2
SHAPE 3
SHAPE 4
SERIES TRIM TRANSDUCER
PS * 1
PS 3
EMERGENCY
STANDBY
PRIMARY
PS 2
PS 4
INVERTER
SAS SENSORS AND SHAPING ASE
AIRCRAFT ELECTRICAL 1
AIRCRAFT ELECTRICAL 2
BATTERY OR APU
1
2
3
4
* D1
D2
D3
D4
1
2
3
4
SHAPE 1
SHAPE 2
SHAPE 3
SHAPE 4
COLLECTIVE TRANSDUCER
FAILURE DISPLAY

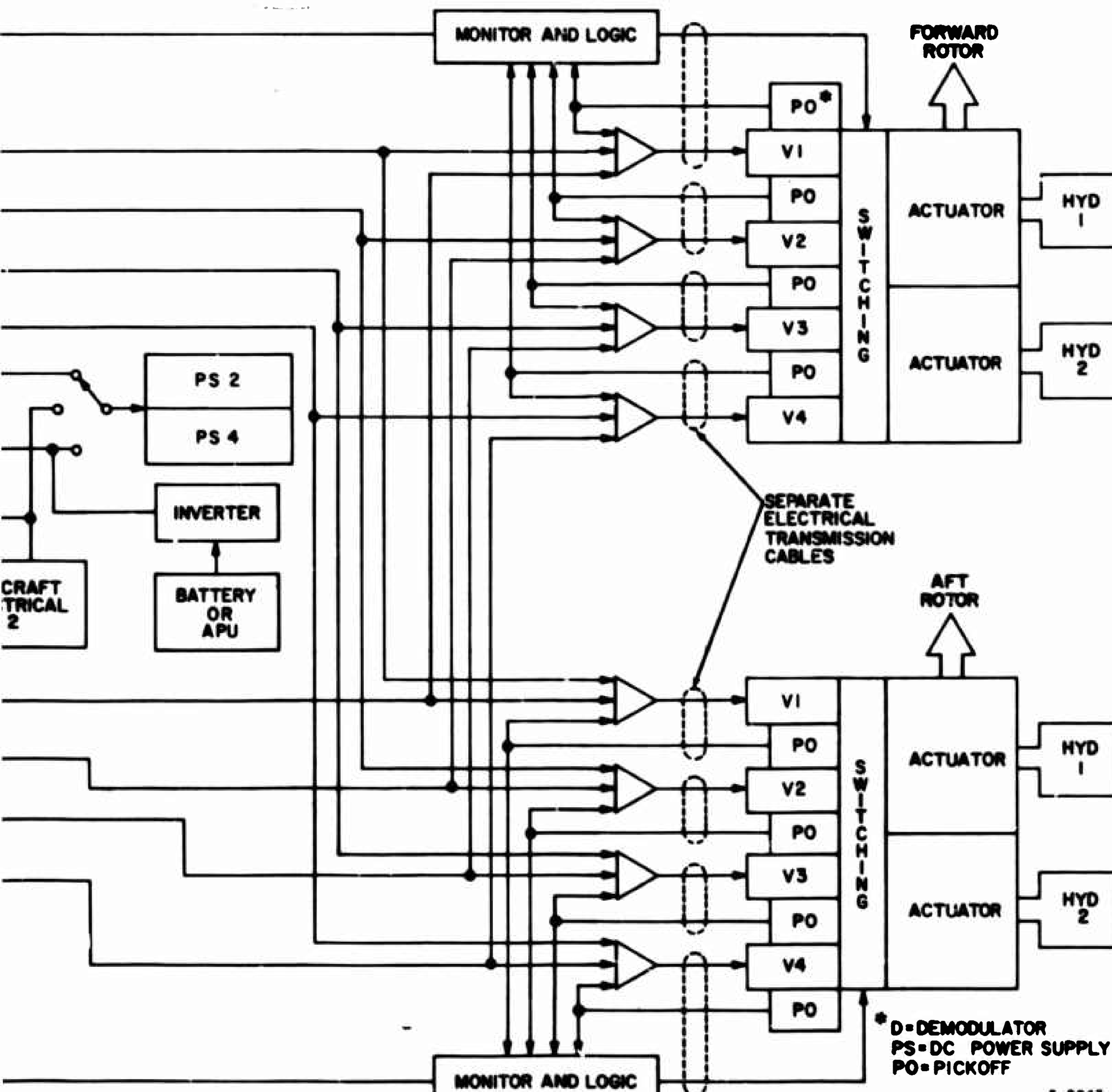


Figure 91
Fly-By-Wire Implementation
Longitudinal Axis

Substituting the estimated failure rates

$$P_f = 2 \times 4\left[2(195 + 27.6)10^{-6}\right]^3 + 4(75 \times 10^{-6})^2$$

$$P_f = 7 \times 10^{-10} + 224 \times 10^{-10}$$

$$P_f = 2.31 \times 10^{-8}$$

Our reliability criterion is $P_f = 1.5 \times 10^{-7}$ each hour for two axes.

Therefore, the reliability of the fly-by-wire system exceeds the required value by a factor of six.

Mean time between maintenance action (MTBM) is calculated directly as the inverse of the total failure rate of the system. The maintenance time for each flight hour is estimated on the basis of the time to replace similarly packaged equipment in other aircraft The mean corrective maintenance time is based on the estimated time required to confirm a reported fault, isolate it to the appropriate replaceable assembly, remove the failed assembly, replace it with a known good one, and retest the system to verify correction of the fault. Fault isolation and checkout after correction uses only built-in test equipment at the flight line level. Maximum time to isolate 95 percent of the potential faults to a replaceable assembly is 2 minutes (estimated). Scheduled replacement of components/equipment on a predetermined basis is not assumed. The predicted mean maintenance time for each flight hour is approximately 2 minutes (table XVIII). From Vertol's maintenance records, the maintenance time for each flight hour on the CH-46 control system is about 30 minutes as compared to approximately 4 minutes for a fly-by-wire system.

TABLE XVI

SUMMARY OF ESTIMATED COMPONENT FAILURE RATES

Component	Number For Each System	Individual Failure Rate Each 10^6 Hours	Total Rate Each 10^6 Hours
Position Transducer (Linear Synchro)	16	10	160
Electronics			
Demodulator	32	2	64
Summing Amplifier	32	1.5	48
Shaping Network	16	2	32
Servo Amplifier	16	4	64
Comparator	24	4	96
Logic Net	4	2	8
Power Supply	8	8	64
Connector	44	0.2	9
Actuator			
Single-Stage Jet-Pipe Valve	16	150	2,400
Actuator	16	35	560
Main Spool Valve	8	40	320
Resolver	16	10	160
Total			3,985

TABLE XVII

SUMMARY OF FLY-BY-WIRE SYSTEM RELIABILITY

Components	Number For Each Channel	Failure Rate (λ) Each 10^6 Hours For Each Channel
Electronic Channel		
Linear Synchro	1	10
Demodulator	2	4
Summing Amplifier	2	3
Shaping Network	1	2
Servo Amplifier	1	4
Power Supply	0.5	4
Connector	2.75 (average)	0.6
Total		27.6

TABLE XVII (cont)

SUMMARY OF FLY-BY-WIRE SYSTEM RELIABILITY

Components	Number For Each Channel	Failure Rate (λ) Each 10^6 Hours For Each Channel
Servo Channel		
Single-Stage Valve	1	150
Actuator	1	35
Resolver	1	10
Total		195
Boost Actuator		
Main Spool Valve	1	40
Actuator	1	35
Total		75

TABLE XVIII

MAINTAINABILITY ESTIMATE

Line Replaceable Assembly	N Number For Each System	Failure Rate Each 10^6 Hours	MTTR Mean Time to Replace at Line (minutes)	MTR Mean Time* to Repair (minutes)
Stick Position Transducer	4	40	30	30
Electronic Assembly	4	96	10	60
Actuator	4	860	30	240

*Includes shop repair and maintenance overhaul.

Maintenance time for each flight hour: $\frac{MT}{FH} = N\lambda \frac{\frac{(MTTR + MTR)}{OH}}{\left(\frac{FH}{OH}\right)}$

where OH/FH is the number of operating hours for each flight hour, which is assumed to be two.

$$\frac{MT}{FH} = \frac{4(40)\ (30 + 30)}{(10^6)(\frac{1}{2})} + \frac{4(96)(10 + 60)}{(10^6)(\frac{1}{2})} + \frac{4(860)\ (30 + 240)}{(10^6)(\frac{1}{2})}$$

$$\frac{MT}{FH} = \frac{8}{10^6} \quad (2400 + 6720 + 232{,}200)$$

$$\frac{MT}{FH} = 1.92 \frac{\text{minutes}}{\text{flight hour}}$$

$$MTBM = \frac{1}{\Sigma\lambda} = \frac{1}{3985 \times 10^{-6}} = 250 \text{ hours}$$

SECTION IX

CONCLUSIONS AND RECOMMENDATIONS

1. CONCLUSIONS

One of the major problems facing aircraft designers is that simple direct mechanical linkages, cables, and feel springs for manual control cannot meet the greater demands of advanced aircraft control system design requirements. These complex manual control systems have increased requirements for space and weight in aircraft where both are at a premium. Nonlinearities such as deadband, hysteresis, and backlash result from the increased compliance, inertia, and friction of complex mechanical devices. These nonlinearities degrade the performance of the control system, and as a result, the full capabilities of the aircraft are not realized. Additional control problems also result from temperature variations and airframe flexibility. The solution to these problems is to replace the mechanical system with a fly-by-wire system. The problem then remains to establish the design technology and criteria for practical fly-by-wire systems within the present state of the art in control systems design and components.

A fly-by-wire flight control system is an electrical primary flight control system employing feedback such that vehicle motion is the controlled parameter. No mechanical backup is used. Fly-by-wire provides a redundant integrated flight control system with the reliability and flexibility necessary to solve the increasingly complex problems of flight controls. The analysis in this report shows that additional benefits are also obtained.

a. Reduces overall weight by 150 to 700 pounds or more

b. Recovers a major part of the volume allowed for control linkage and cable motion thereby increasing the usable space for other subsystems

c. Improves control performance by eliminating friction, inertia, backlash, compliance, and the effects of temperature and body bending

d. Reduces initial controls design effort and simplifies installation and maintenance

e. Provides feasibility of standardizing flight control systems between aircraft and increases flexibility of cockpit installation

f. Decreases vulnerability especially if self-contained actuators are employed

To meet the control system reliability requirements, fly-by-wire controls employ redundancy techniques. Redundancy implies an increase in the number of components which would appear to imply an increase in the number of maintenance actions required. Modular packaging and failure reporting circuits keep maintenance time low, however.

The study has determined the fly-by-wire system requirements, established the design criteria, developed an approach to system design, and established the availability of the components required. The primary requirement of the flight control system is to allow the pilot to control the flight path of the vehicle with a minimum of effort and error. Hence, the system includes not only the link between the control stick and control surface but the artificial feel system and trim as well.

The fly-by-wire system includes the following functional elements.

a. A spring-centered control stick (or wheel) with an electrical output proportional to force and related to position. The stick may be located at the center or to one side of the cockpit

b. C* feedback ($C^* = K_1 n_z + K_2 \dot{\theta} + K_3 \ddot{\theta}$) in pitch and angular rate in roll and yaw to provide a closed loop control system around the airplane. This provides good artificial feel and path control independent of airspeed and altitude and the aircraft type (within a given class of aircraft)

c. Electronics for signal shaping, summing, switching, gain control, and monitoring

d. Control actuators which combine the functions of the power actuators and the series trim and stability augmentation actuators

e. Trim

f. Power sources

g. Failure display panel

The study has established the following design criteria and requirements for fly-by-wire control system.

a. The system must remain operational after any two failures with a probability of system failure for a 1-hour flight no greater than 2.3×10^{-7}.

b. Control channels from stick to surface must be quadruplex; this can mean three real channels and a model or four real channels. The power sources must be at least independent triplex and monitored. The C* sensors must be at least triplex with self-test capability.

c. The system should be able to operate on unregulated and unsynchronized power.

d. Performance must be undegraded after the first failure; have limited degradation after a second failure in the same axis; and fail to neutral or a preselected trim position upon a third failure. Allow the pilot to select any channel after a third failure in the same axis.

e. Monitor points should be on actuator rate, at the C* signal just before summing with the command signal, and/or just before the servo actuator input.

f. Failure reporting is required to inform the air and ground crews of the control system's status. Failure isolation to a line replaceable unit is required to minimize maintenance time.

g. Preflight self-test to check system integrity should take no more than 30 seconds.

h. Channel transfer time in case of failure should be kept to a minimum (less than 50 milliseconds for the application of this study).

i. Electronics should be packaged in potted modules by channel and physically isolated to minimize damage effects.

j. Electrical cables should be protected by conduit where necessary and be separately routed to minimize damage effects.

The components chosen for use in fly-by-wire systems are of proven design and essentially off-the-shelf with the exception of the actuators. Even here the existing technology is employed, but configurations had to be found that were compatible with the higher degree of redundancy and transfer time requirements. The following types of components were selected.

a. Position transducers are inductive and brushless, such as the LVDT (linear variable differential transformer), induction potentiometer, microsyn, and E-core variable transformer. These include control stick, actuator feedback, and C* sensor outputs.

b. Electronics are mixed ac and dc. Signal shaping and summing must be dc to avoid power phasing problems. Fail-passive design requires ac circuitry. Both microcircuits and discrete components are used.

c. Conventional rate gyros and direct-measuring normal accelerometers incorporating self-test capabilities are used.

d. Reinforced, stranded copper wire (shielded where necessary) should be used for added strength.

e. The fail-passive secondary actuator was selected because it has a number of important advantages over the other candidate actuator configurations. The advantages include: no failure transfer time, fewest number of moving parts, the most tolerant of channel mismatch and dirty hydraulic fluid, and the lightest and least expensive design.

Based on the results of the study, our conclusions are that fly-by-wire control provides many advantages over mechanical flight control systems and that control system technology has reached the point where practical fly-by-wire system designs can be realized today. The main obstacle fly-by-wire has to overcome is the lack of confidence in system integrity caused by a distrust of nonmechanical system reliability.

2. RECOMMENDATIONS

To overcome the lack of confidence, an existing aircraft, particularly one with known control system problems, should be converted to fly-by-wire control and flown to demonstrate its feasibility. Then, many flight hours will provide in-flight proof of its maturity and safety. The work should progress in two phases. The first phase should be to build an experimental laboratory model of a fly-by-wire system to demonstrate its operation under simulated failures. This model would also demonstrate the use of state-of-the-art components and the effectiveness of existing design techniques.

Construction and evaluation of a laboratory experimental model of a representative fly-by-wire system would accomplish a number of ends. First, it would demonstrate the system's operation and performance under various failure conditions; second, it would provide data to establish performance requirements; third, it would provide a test bed for testing other techniques which may become available during the course of the program; and fourth, it would provide the data needed to design future flightworthy systems.

The second phase should be to convert an existing aircraft to fly-by-wire using the data from the experimental model. Initial system tests should be run with the electrical system paralleling the existing mechanical system in the airplane with reversion to the mechanical system available for emergencies. The initial tests would uncover any problems which may exist as they might in any new system. After flying the system for a specified number of hours without using the mechanical reversion capability, remove the mechanical system so that the fly-by-wire system can continue on its own to build up flight time. Then, by putting as many flight hours on it as possible, the integrity and practicality of fly-by-wire can be demonstrated. Final proof of the fly-by-wire feasibility can only come in this way. Successfully completing this step should provide the impetus to all those people in industry who are waiting for in-flight proof of fly-by-wire maturity.

SECTION X

GLOSSARY OF TERMS

ACHIEVED RELIABILITY - The reliability demonstrated at a given point in time under specified conditions of use and environment.

ACTIVELY REDUNDANT SYSTEMS - Several identical channels or elements which perform normal control functions simultaneously and identically at all times.

ARTIFICIAL FEEL SYSTEM - A group of components operating in conjunction with the pilot's input to the control stick to artificially provide the pilot with control pressure cues corresponding to the aerodynamic forces and responses of the vehicle.

AVAILABILITY - The fraction of the total desired operating time that material actually is operable.

BREADBOARD MODEL - An assembly of preliminary circuits and parts to prove the feasibility of a device, circuits, equipment, system or principle in rough form, without regard to the eventual overall design or form of parts.

C* - A blend of pitch attitude rate (radians/second) and acceleration (radians/second^2) and normal acceleration (feet/second^2) according to the expression, $C^* = K_1 n_z + K_2 \dot{\theta} + K_3 \ddot{\theta}$

CATASTROPHIC FAILURE - A sudden change in the operating characteristics of materiel resulting in a complete lack of useful performance.

CLOSED CENTER VALVE - A valve in which the output spool lands completely overlap the valve ports when the valve is nulled or centered.

COMPONENT - A functional part of a subsystem or equipment which is essential to operational completeness of the subsystem or equipment, and which may consist of a combination of parts, assemblies, accessories, and attachments.

CONFIDENCE FACTOR - The percentage figure that expresses confidence level.

CONFIDENCE INTERVAL - A range of values which is calculated from the data so as to have a given probability (confidence level) of containing the true value of the universe characteristics.

CONFIDENCE LEVEL - The probability that a given statement is correct, or the chance that the true value lies between two confidence limits (the confidence interval).

CONTROL STICK TRANSDUCER - A component attached to the control stick which converts either the stick displacement or the force applied by the pilot to the stick into a proportional signal, usually electrical.

DEBUGGING - A reliability conditioning procedure which is a method of aging the equipment by operating it under specified environmental and test conditions in accordance with an established procedure to eliminate early failures and age or stabilize the equipment prior to final test and shipment. Sometimes called "burn-in".

DITHER - A low amplitude sinusoidal signal superimposed on the input signal to the servovalve to reduce "stiction" and deadzone.

DOWN TIME - The total time during which the system is not in condition to perform its intended function. (Down time can in turn be subdivided in the following categories: repair time, logistic time and administrative time.)

EARLY FAILURE PERIOD - That period of materiel life starting just after final assembly where failures occur initially at a higher than normal rate due to the presence of defective parts or abnormal operating procedures.

ELECTRICAL PRIMARY FLIGHT CONTROL SYSTEM - A flight control system mechanization wherein the pilot's control commands are transmitted to the moment or force producer only via electrical wires.

ELECTROMECHANICAL ACTUATOR - a mechanism for converting electrical energy into mechanical motion, usually rotary. Also, electromechanical servomotor.

EXPERIMENTAL ENGINEERING MODEL - Components and/or devices which may be either actual or simulated and represent a model of some particular design whereby system performance can be ascertained by application and instrumentation of input-output relationships.

FAIL-ACTIVE - A failure condition in which a failed channel or element interferes with the normal operation of a redundant channel or element.

FAIL NEUTRAL - A mode of operation of the flight control system whereby the flight controls assume a neutral point after failures have either put, or tended to put the aircraft in an unsafe flight condition.

FAIL-OPERATIONAL - A characteristic of a system in which normal system operation is maintained after a single failure. During the failure occurrence, and during detection and switching sequence (if required), the aircraft shall not be placed in an unsafe or unrecoverable position. In commercial aircraft, passenger comfort and awareness must also be considered.

FAIL-PASSIVE - (1) A failure condition in which a failed channel or element cannot interfere with the normal operation of a redundant channel or element; (2) A system in which, due to a failure, system operation is lost without a significant output to interfere with operator (i.e., pilot) takeover.

FAIL-SAFE - A system failure condition similar to fail-passive, in which small transients are allowed as long as the aircraft is not placed in an unsafe attitude. It is a term related only to the application of a system and not germane to the classification of redundancy techniques.

FAIL-SOFT - Fail-safe.

FAILURE - The inability of materiel to perform its required function within previously established limits.

FAILURE - DEPENDENT (Secondary) - One which is caused by the malfunctioning of associated items. Not independent.

FAILURE DETECTING SYSTEMS - Determine that a failure has occurred by using auxiliary equipment, and switch out the failed equipment to achieve the required fail-passive or fail-operative condition of the failed channel or element.

a. Operator Selection - Uses the operator (i.e., pilot) to determine a failure and to perform switching required.

b. Channel Monitor - Uses a comparison between an operating channel and a second channel (real or simulated) to determine a failure and to initiate any switching required to achieve a fail-passive condition.

c. Shared Monitor with Voting - Uses disagreement with the majority of a number of channels or elements to determine a failure and thereby to initiate switching required to achieve a fail-passive condition.

d. Self-Test - Uses a test signal through the channel or element whose output is compared with a standard to determine a failure and thereby to initiate switching required for the fail-passive condition of the failed channel or element.

FAILURE - INDEPENDENT (Primary) - One which occurs without being related to the malfunctioning of associated items. Not dependent.

FAILURE MODE - The physical description of the manner in which a failure occurs. Also, in analysis of design reliability, a description of the manner in which an equipment function may be affected by a failure.

FAILURE RATE - At any point in the life of materiel, the incremental change in the number of failures per associated incremental change in the measure of life (cycles, time, miles, events, etc, as applicable).

FEEDBACK TRANSDUCER - The component of a closed-loop control system or servomechanism which converts the output into a related signal, usually electrical. A proportionality factor (e.g., volts/radian) specifies the gain as required for the design of the servomechanism.

FLY-BY-WIRE - An electrical primary flight control system employing feedback such that vehicle motion is the controlled parameter.

GAUSSIAN OR NORMAL DISTRIBUTION - A density function of a population which is bell shaped and symmetrical, and which is completely defined by two independent parameters, the mean and the standard deviation.

GROUP REDUNDANCY - Redundancy applied at the component or subsystem level.

HARDOVER FAILURE - A failure of a device such that the device's output quantity goes to its maximum value.

HERTZ - Cycles per second.

HUMAN FACTORS - Facts about human behavior which affect the design of systems. As a discipline, its goal is to achieve an optimal system with an efficient man working in a safe and habitable environment. Such a man will be working with equipment designed to maximally use his capabilities, and minimize his limitations, while reliably performing tasks that men can do better or more economically than machines.

HYDRAULIC ACTUATOR - A device for converting fluid energy into mechanical motion.

HYDRAULIC SERVOACTUATOR - A mechanism consisting of a servovalve and a linear actuator.

HYDRAULIC SERVOVALVE - A mechanism which converts some form of input signal (electrical or mechanical) into a proportional fluid flow or pressure which may be used to move an actuator.

HYSTERESIS - Valve hysteresis is the difference in input signal necessary to produce flows increasing from zero to maximum from that necessary to produce flows decreasing from maximum to zero.

IMPORTANCE FACTOR - The ratio of the number of mission failures caused by materiel failing to the total number of failures of the materiel. The relative importance of the particular materiel to the total mission effectiveness.

INDIRECT MONITORING - Monitoring of components or subsystems for proper operation to determine (by inference) that the system is operating properly.

INHERENT RELIABILITY - The reliability potential present in the design.

JET PIPE - A device sometimes used to provide a velocity head for controlling the position of an output stage. It consists of a rotatable nozzle which proportions flow between two adjacent holes which lead to opposite ends of the output member.

LARGEST VALUE SELECTOR - Selects and transmits the input signal with the largest magnitude; independent of input signal sign.

LIFE CHARACTERISTICS - Failure rate plotted as a function of the measure of life (cycles, time, miles, events, etc , as applicable).

LINEAR ACTUATOR - Also a cylinder or ram. An actuator consisting of a movable element such as a piston or ram operating within a cylinder bore. The actuator may be single ended or double ended.

MAINTAINABILITY - The quality of the combined features of equipment, design and installation which facilitates the accomplishment of inspection, test,

servicing, repair, and overhaul with minimum time, skill, and resources in the planned maintenance environment.

MAJORITY VOTE LOGIC - A binary operator which selects the signal which agrees with the majority of three or more signals.

MEAN-TIME*-BETWEEN-FAILURES - For a particular interval, the total measured functioning time* of a population of materiel divided by the total number of failures within the population during the measured period.

MEAN-TIME*-TO-FAILURE. - The mean functioning time* at which the first failure materiel becomes expended.

MEAN-TIME*-TO-FIRST-FAILURE - The mean functioning time* at which the first failure occurred.

MECHANICAL CONTROL SYSTEM - A manual flight control system in which the means of control between the pilot's station and the control actuators is through mechanical linkages or cables.

MID-VALUE LOGIC - An analog operator which selects the signal having the middle value of three or more signals.

MISSION RELIABILITY - The probability that the materiel will give specified performance for the duration of a mission when used in the manner and for the purpose intended, given that the materiel is functioning properly at the start of the mission.

MODULE - A combination of components, contained in one package or so arranged that they are common to one mounting, which provides a complete function or functions to a system and/or subsystem in which they operate.

MONITOR - The combination of a model or real channel or element and a comparator.

a. Model is a dynamic simulation or a stored transfer matrix of the characteristics of a real channel or element.

b. Comparator is a circuit whose output reports agreement or disagreement between several channels, or between channels and a model.

NONFAILURE DETECTING SYSTEMS - Achieve fail-passive or fail-operational performance without failure detection or switching.

a. Massive Redundancy uses sufficient identical functional blocks with a parallel output scheme so that a failure of one equipment can be ignored since it provides a small part of the total or composite system output.

b. Inherent Fail-Passive pertains to channels or elements which are fail-passive without the addition of failure detecting or logic equipment.

c. Logical Selection uses a logical basis for selecting an output, from several channels or elements, which is most likely to be a satisfactory signal for the system output. Included here are mid-value logic and self-organizing systems. A system may use several different redundancy techniques in different parts to achieve fail-operational capability.

NORMAL DISTRIBUTION - (See Gaussian distribution.)

NOZZLE FLAPPER - A device often used as the first stage to proportion the pressure across the output spool. This is accomplished by varying the area of the variable orifice with respect to a fixed orifice by varying the distance of the flapper from the nozzle which passes the fluid.

OPERATING TIME - The time period during which the materiel is performing its intended function.

PRESSURE CONTROL VALVE - A valve in which the output load pressure is approximately proportional to input differential current irrespective of flow.

PREVENTIVE MAINTENANCE - A procedure of periodically checking and/or reconditioning a system to prevent or reduce the probability of failure or deterioration while in service.

PSEUDO-FLY-BY-WIRE - A Fly-By-Wire flight control system with a normally disengaged mechanical backup.

PREVENTIVE MAINTENANCE - A procedure of periodically checking and/or reconditioning a system to prevent or reduce the probability of failure or deterioration while in service.

RANDOM FAILURE - Any failure whose occurrence is unpredictable.

REDUNDANCY - The existance of more than one means for accomplishing a given function.

REDUNDANCY, PARALLEL - The application of two or more means of accomplishing a given task in a system, all of which are functioning at the same time, but each of which is capable of handling the task itself in the event of a failure to the other means.

REDUNDANCY, STANDBY - Redundancy applied to a system to supply an alternate means of accomplishing a task, which is held in abeyance until a failure of the primary equipment is sensed and the standby equipment is actuated to take over the required task.

RELIABILITY - The probability that materiel will perform its intended function for a specified period under stated conditions.

STANDARD DEVIATION - A measure of the spread or dispersion of the distribution of a random variable, mathematically expressed as the positive square root of the second moment about the mean.

SIGNAL TRANSMISSION - The method of coupling the control stick signal to the surface actuator.

SIGNAL TYPE - The format of the control signal; e.g., dc, suppressed carrier ac, pulse width, or digital.

SIMPLE REDUNDANCY - Redundancy applied at the system level.

SINGLE STAGE VALVE - A valve in which the output flow is controlled by a spool directly connected to the electromagnetic driver.

STATUS REPORTING - The process of reporting failures within a system. With a failure detecting system, reporting is a simple indicating process. With a nonfailure detecting system, auxiliary failure detecting equipment is required. However, this equipment does not provide for the fail-operational capability of the system.

SUMMER - The component utilized to algebraically sum signals. Also summing junction.

SYSTEM COMPATIBILITY - The ability of the equipments within a system to work together or perform the intended mission of the system. In a broader sense, system compatibility is the suitability of a system to provide the levels of field performance, reliability and maintainability required by the military services.

THRESHOLD - The minimum amplitude of input signal to a servovalve necessary to obtain measurable flow.

TORQUER MOTOR - An electromagnetic driver which provides a rotary output displacement directly proportional to input current. Generally the arc of rotation is so small that the displacement may be considered linear.

TRADEOFF - The procedure of trading a degree of one attribute to gain a degree of another attribute, e.g., a degree of reliability might be sacrificed to obtain a greater degree of performance under certain conditions, or vice versa.

TWO STAGE VALVE - A valve which contains two stages, the first of which is similar to a single stage valve and which positions a second stage spool hydraulically. The second stage spool then controls output flow.

USEFUL LIFE - The total operating time between debugging and wearout.

VALVE NULL - Refers to the condition where the flow from the output load is zero for zero input current.

WEAROUT - The point at which further operation is uneconomical.

WEAROUT FAILURE - Those failures which occur as a result of deterioration processes or mechanical wear and whose probability of occurrence increases with time.

SECTION XI

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APPENDIX

XB-70A PITCH AND ROLL CONTROL
SYSTEM SCHEMATIC

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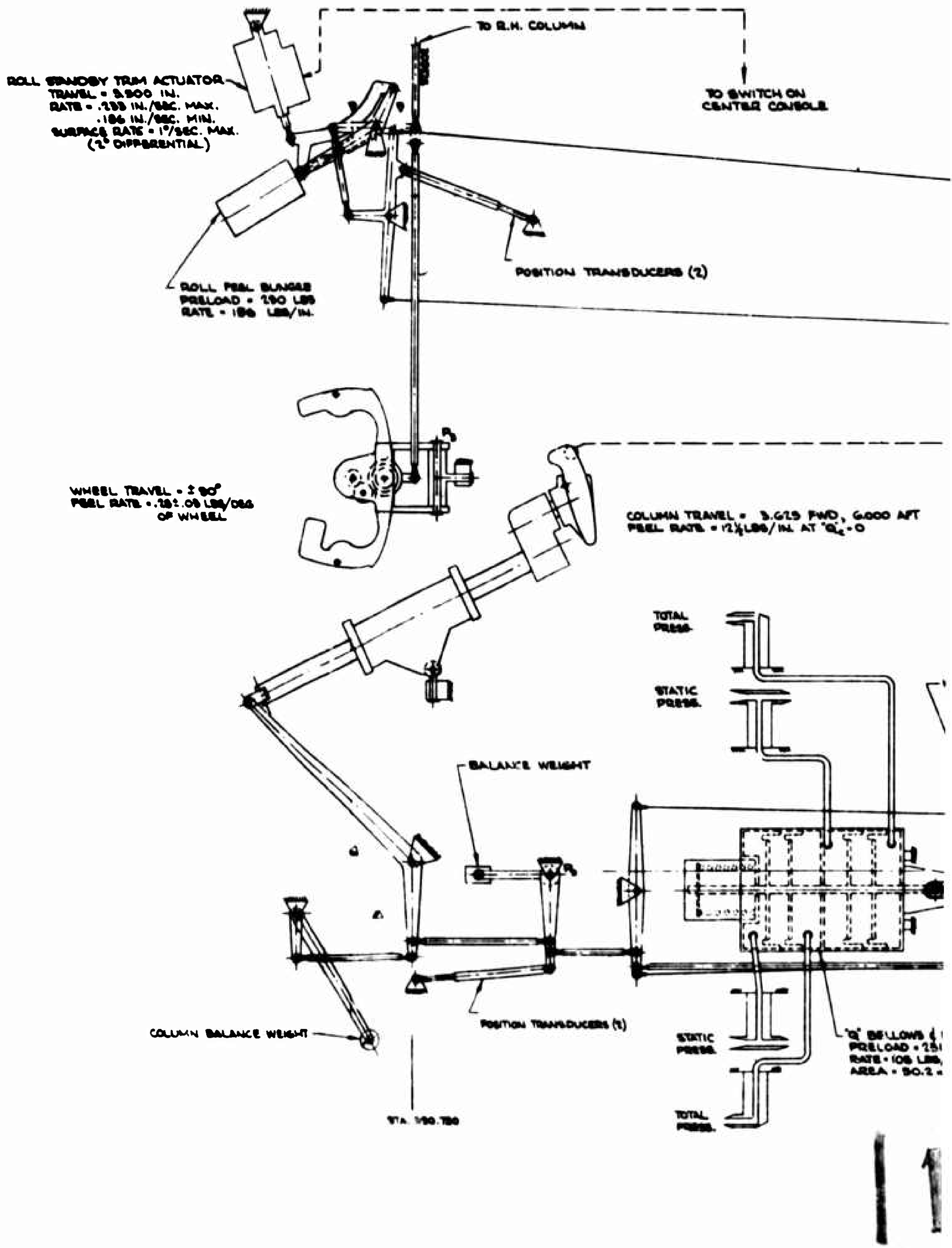

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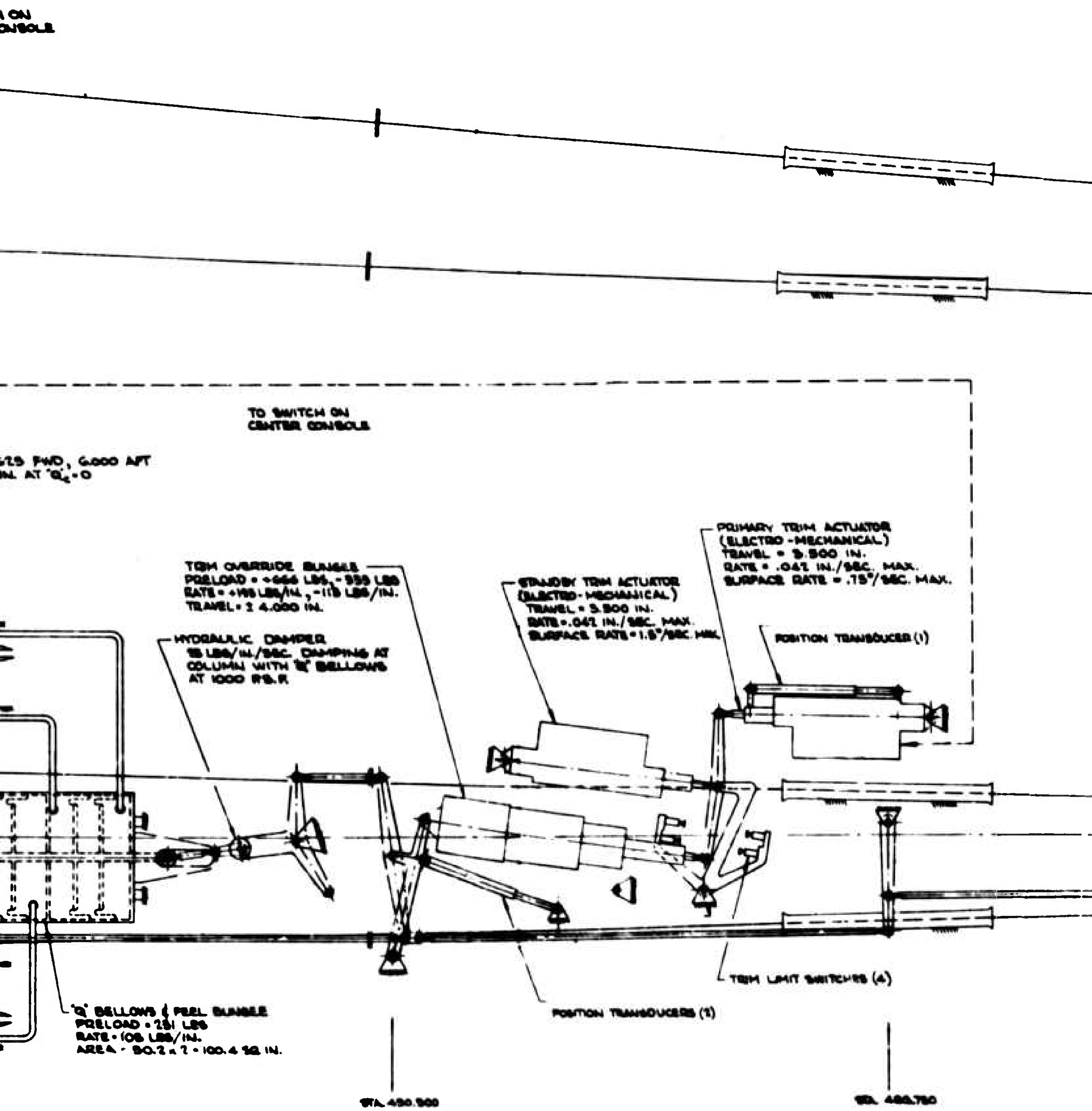
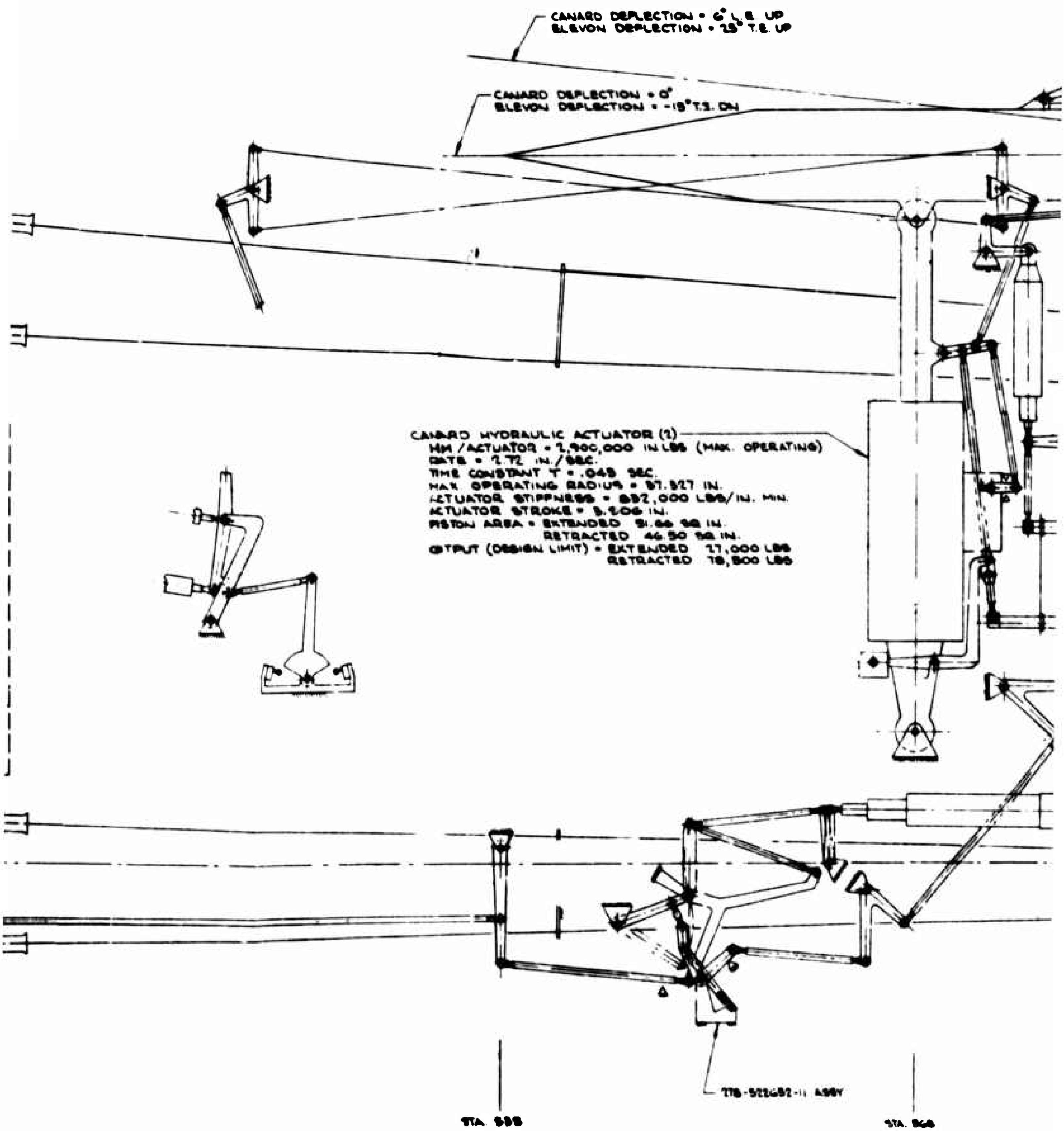




CANARD DEFLECTION = 6° L.E. UP
ELEVON DEFLECTION = 25° T.E. UP
CANARD DEFLECTION = 0°
ELEVON DEFLECTION = -15° T.E. DN
CANARD HYDRAULIC ACTUATOR (2)
HM / ACTUATOR = 2,900,000 IN LBS (MAX. OPERATING)
RATE = 2.72 IN./SEC.
TIME CONSTANT T = .043 SEC.
MAX OPERATING RADIUS = 37.327 IN.
ACTUATOR STIFFNESS = 882,000 LBS/IN. MIN.
ACTUATOR STROKE = 3.206 IN.
PISTON AREA = EXTENDED 51.66 SQ IN.
RETRACTED 46.50 SQ IN.
OUTPUT (DESIGN LIMIT) = EXTENDED 27,000 LBS
RETRACTED 18,500 LBS
STA. 838
STA. 868

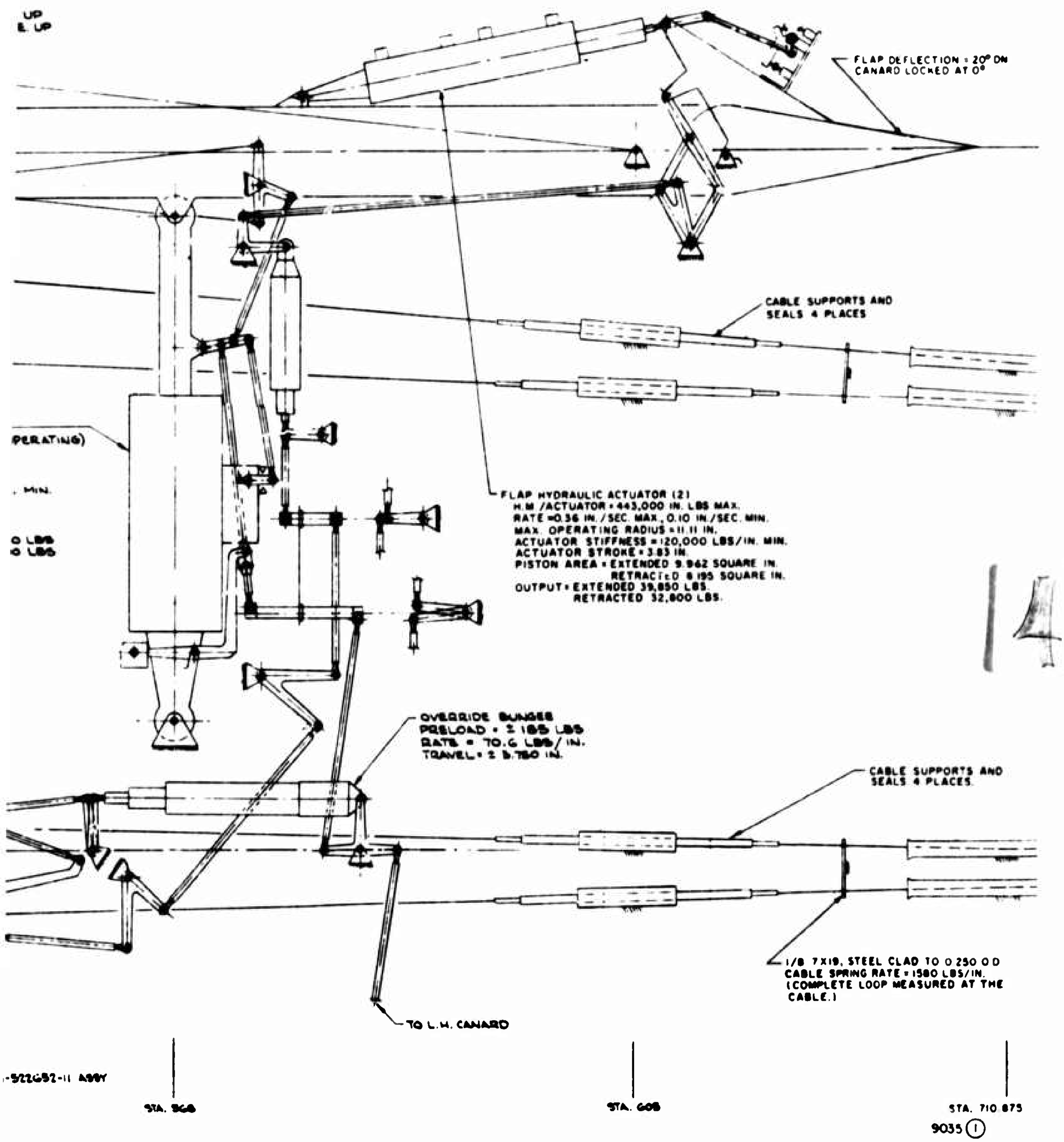
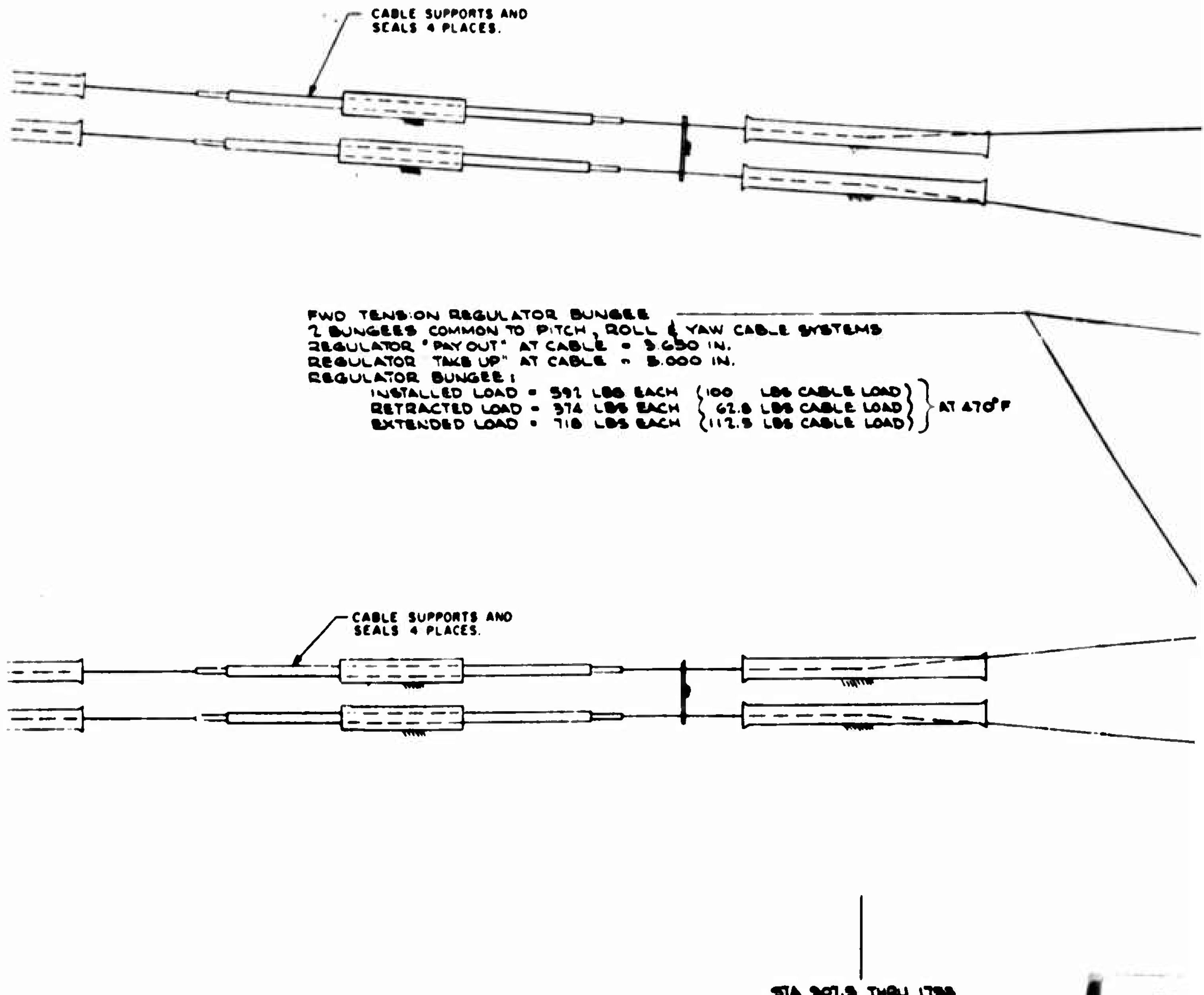
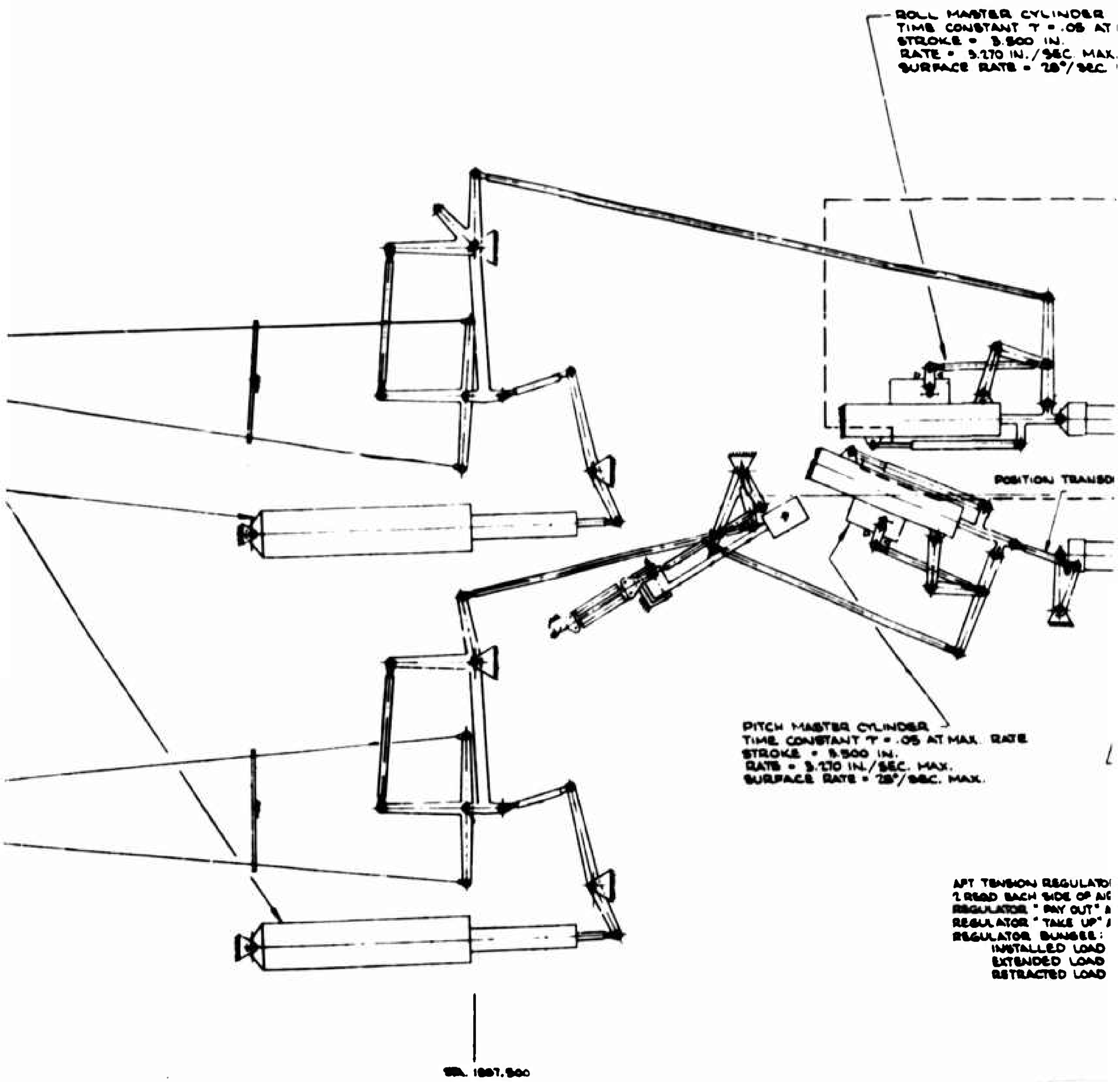


Figure 92
XB-70A Pitch and Roll Control System
Schematic (Sheet 1 of 3)


CABLE SUPPORTS AND
SEALS 4 PLACES.
FWD TENSION REGULATOR BUNGEE
2 BUNGEES COMMON TO PITCH, ROLL & YAW CABLE SYSTEMS
REGULATOR "PAY OUT" AT CABLE = 3.650 IN.
REGULATOR "TAKE UP" AT CABLE = 5.000 IN.
REGULATOR BUNGEE:
INSTALLED LOAD = 592 LBS EACH (100 LBS CABLE LOAD)
RETRACTED LOAD = 374 LBS EACH (62.8 LBS CABLE LOAD)
EXTENDED LOAD = 718 LBS EACH (112.5 LBS CABLE LOAD)
AT 470°F
CABLE SUPPORTS AND
SEALS 4 PLACES.
STA 907.5 THRU 1798


ROLL MASTER CYLINDER
TIME CONSTANT T = .05 AT
STROKE = 3.500 IN.
RATE = 3.270 IN./SEC. MAX.
SURFACE RATE = 28°/SEC
POSITION TRANSD
PITCH MASTER CYLINDER
TIME CONSTANT T = .05 AT MAX. RATE
STROKE = 3.500 IN.
RATE = 3.270 IN./SEC. MAX.
SURFACE RATE = 28°/SEC. MAX.
AFT TENSION REGULATO
2 REQD EACH SIDE OF AI
REGULATOR " PAY OUT "
REGULATOR " TAKE UP "
REGULATOR BUNGEE:
INSTALLED LOAD
EXTENDED LOAD
RETRACTED LOAD
STA. 1887.500

12

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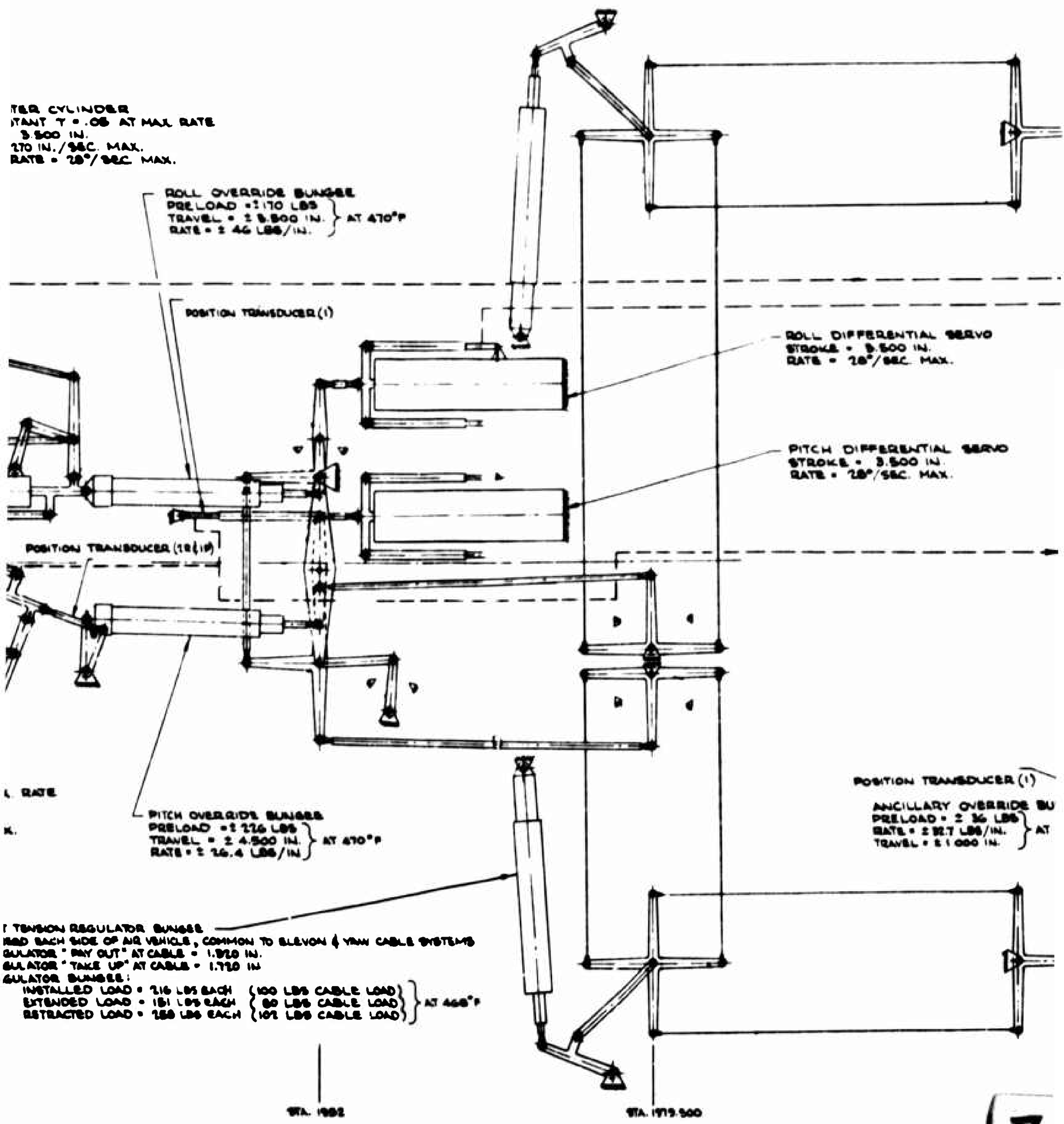

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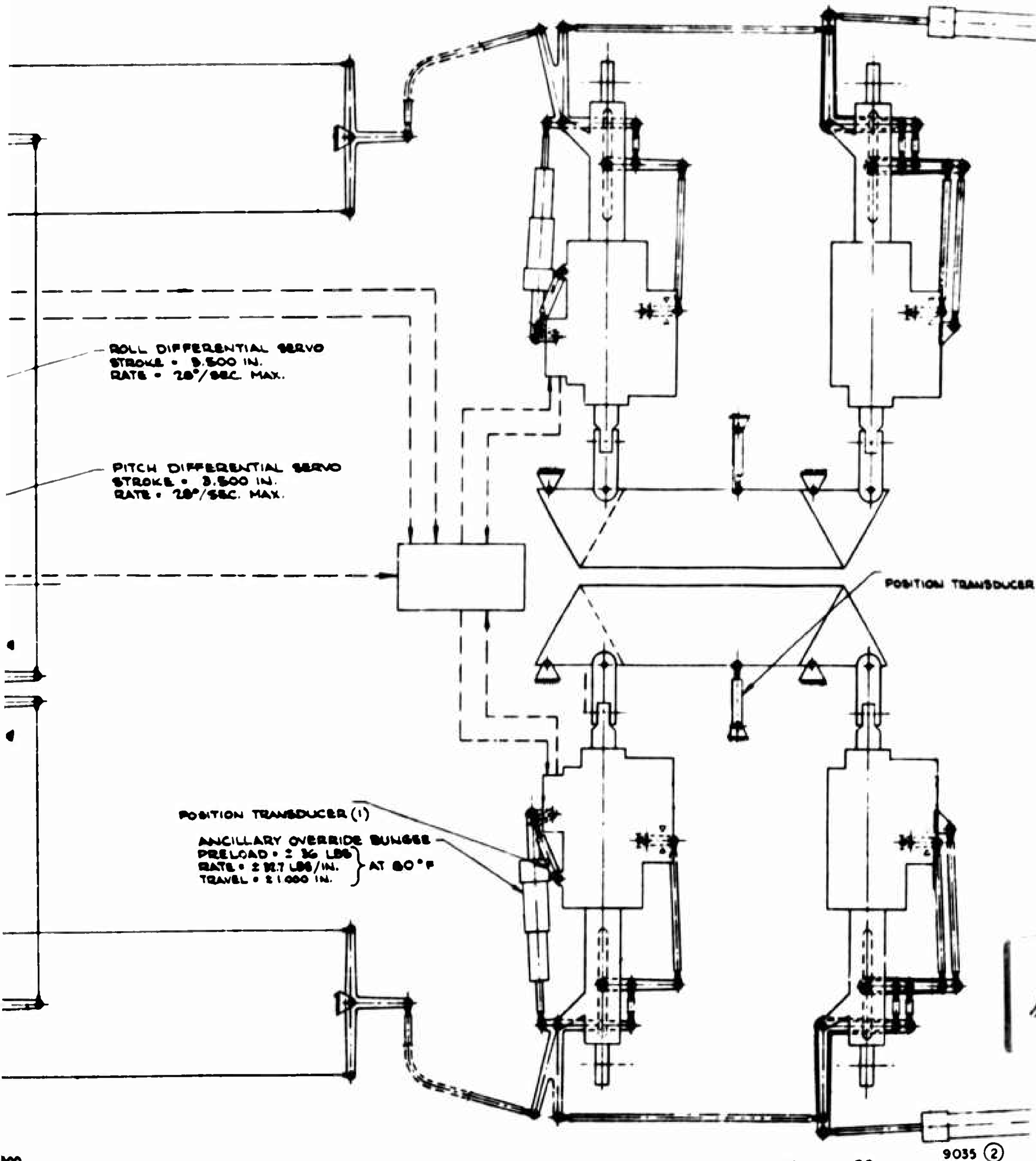

Figure 92
XB-70A Pitch and Roll Control System
Schematic (Sheet 2 of 3)

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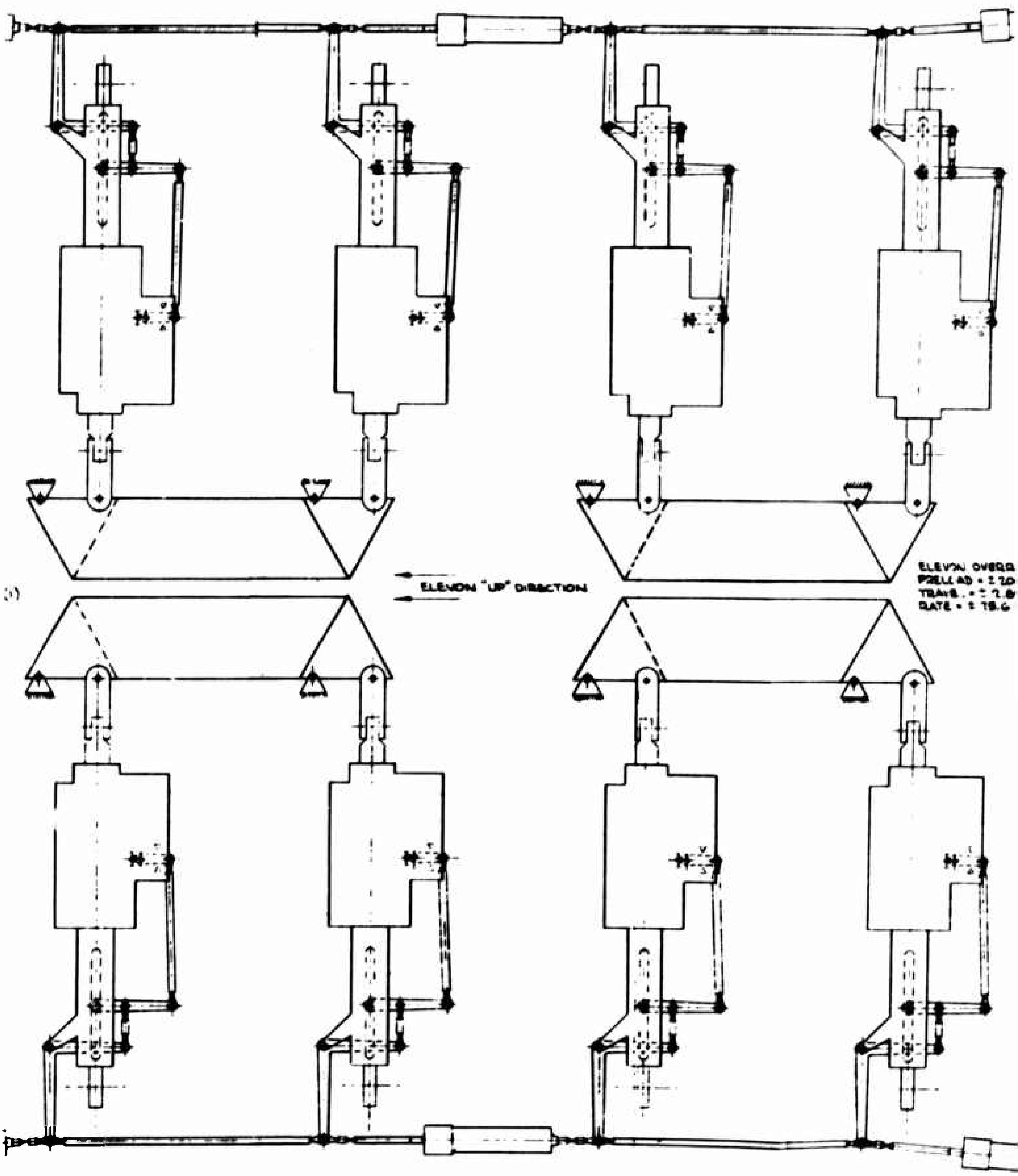

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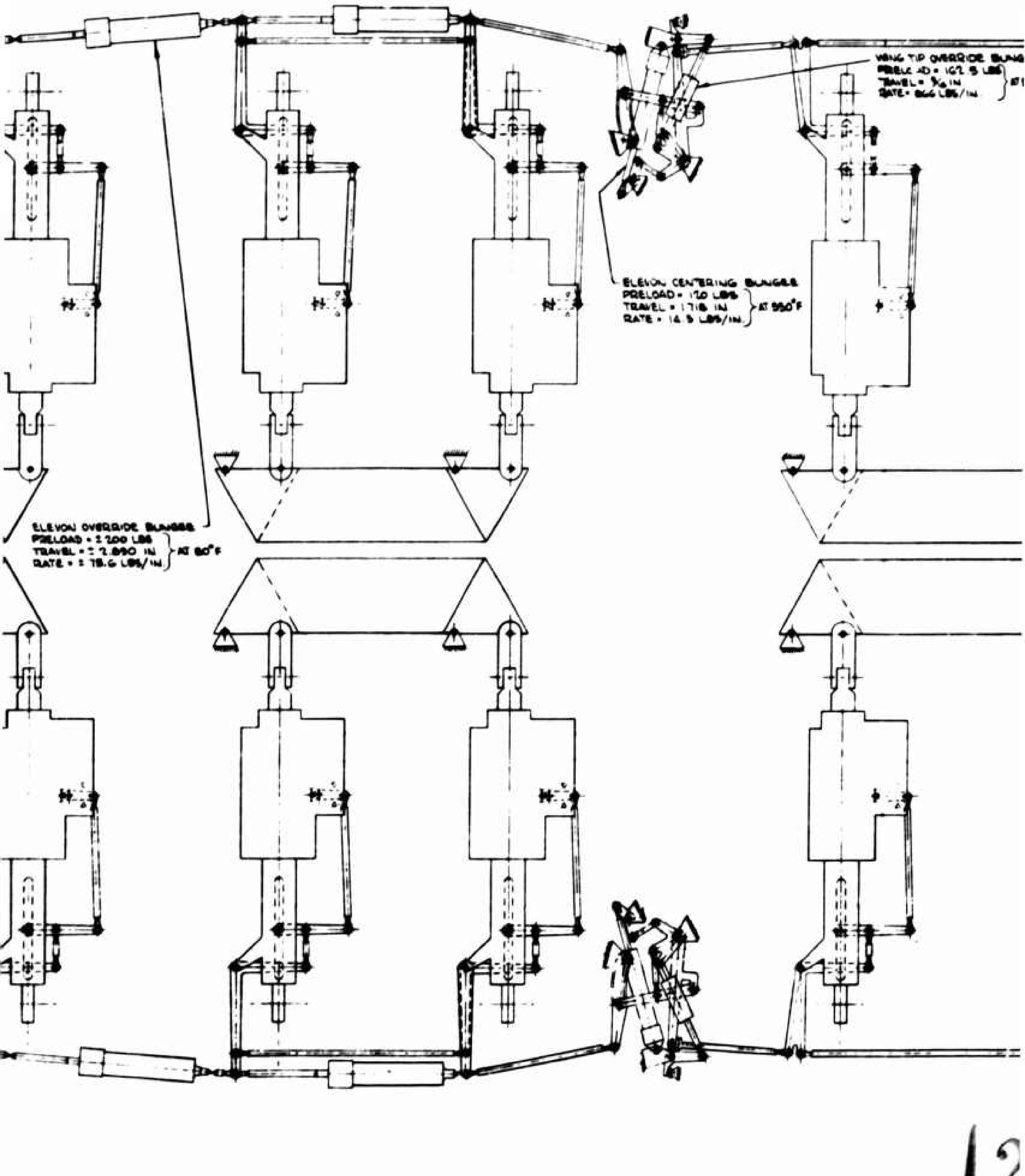

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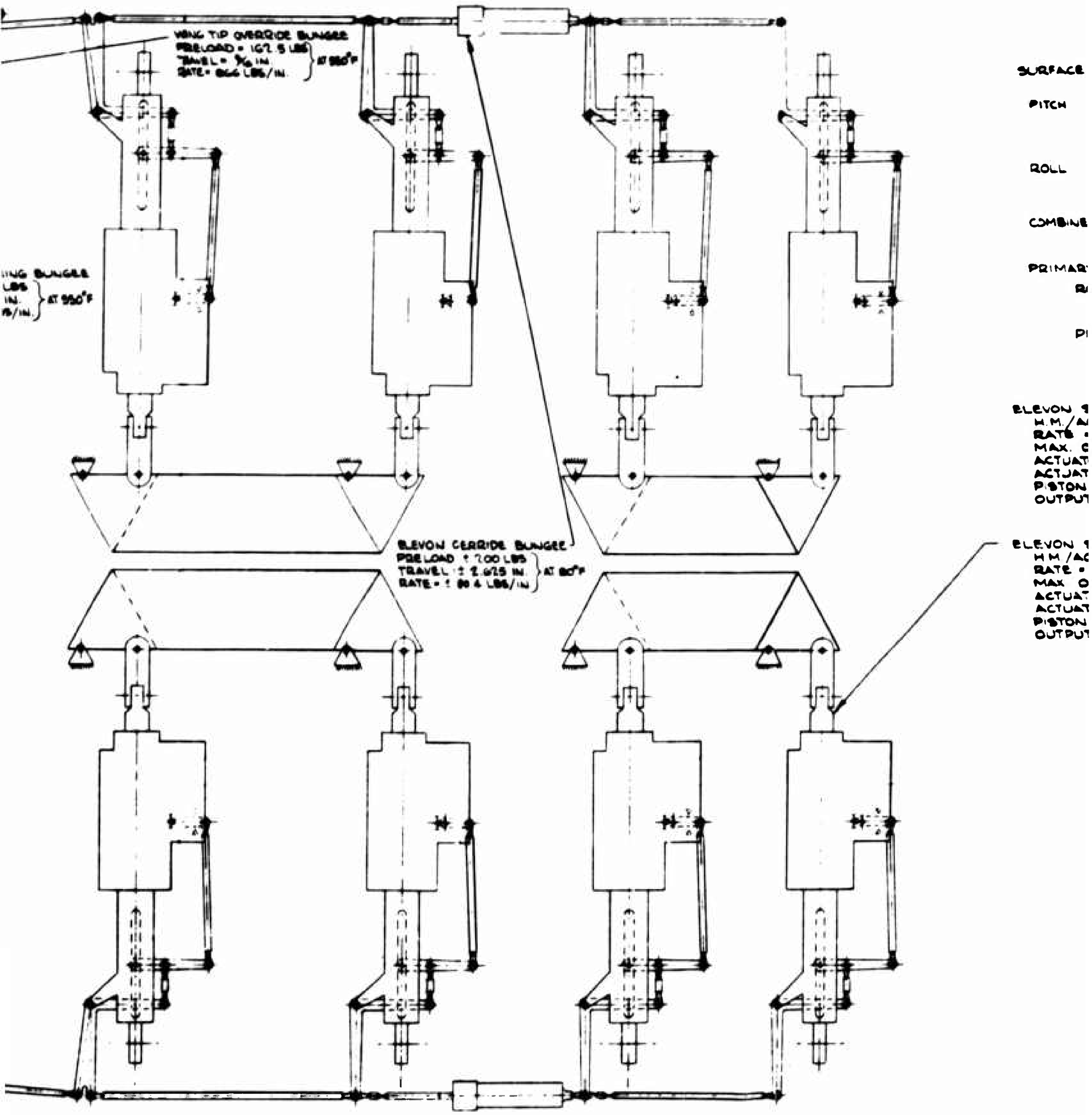

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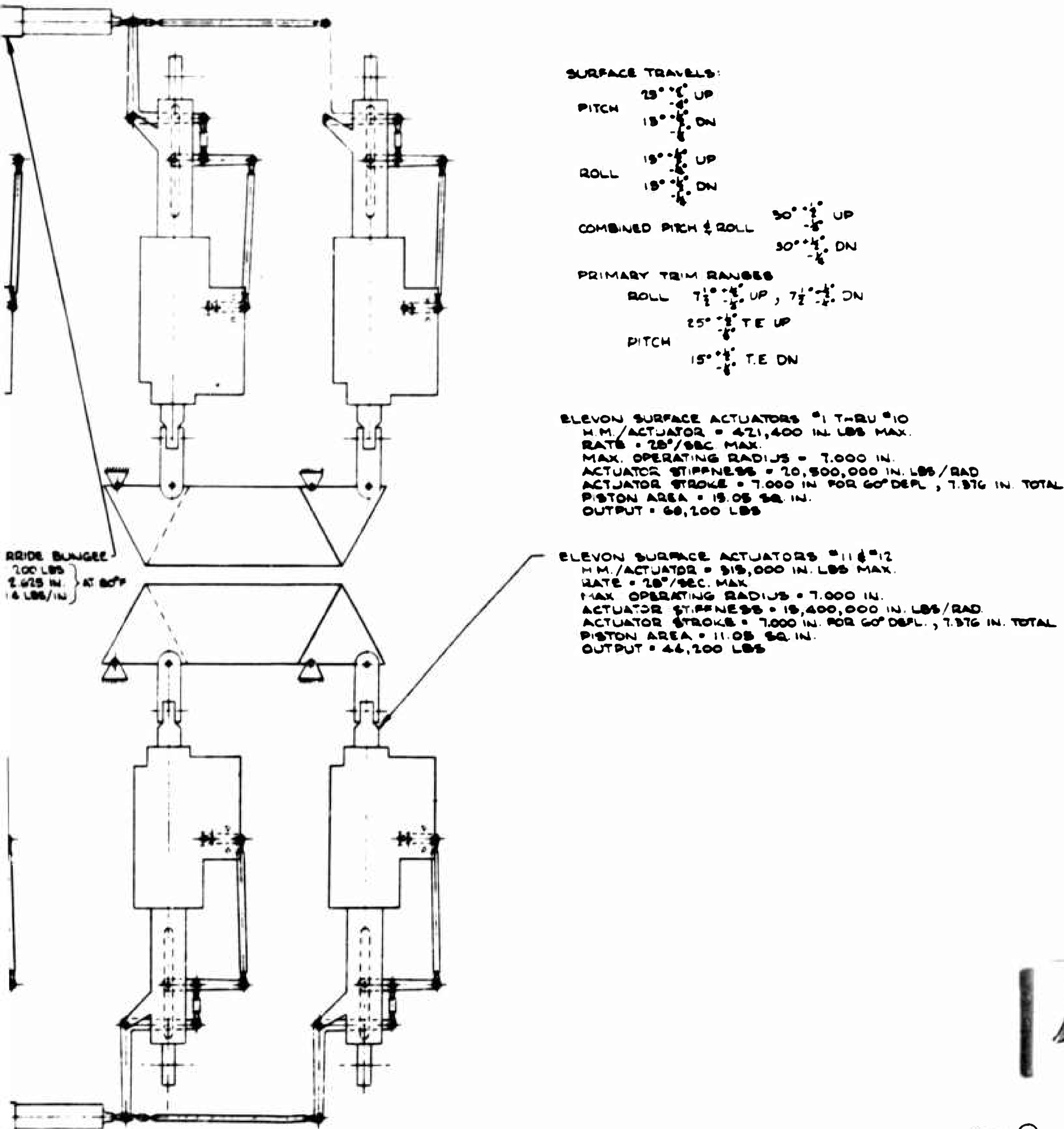

Figure 92
XB-70A Pitch and Roll Control System
Schematic (Sheet 3 of 3)

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11 SUPPLEMENTARY NOTES
This program is related to outline in LJ-PH3845, Study of Research of Fly-by-Wire Techniques.

12 SPONSORING MILITARY ACTIVITY
Air Force Flight Dynamics Laboratory (RTD) Wright Patterson Air Force Base Ohio

13 ABSTRACT
Manual flight control systems are described in which the sole means of control between the pilot's station and the control actuator is in the form of electrical signals. No mechanical control links are used in the system. Such a system is defined as a fly-by-wire control system. Because of the growing number and severity of problems in mechanical control systems, particularly in large and high speed aircraft, fly-by-wire systems are evolving out of necessity. Fly-by-wire control is shown to provide many advantages over conventional mechanical flight control systems. Principally, they are reduced weight and volume, improved control performance, reduced design effort and maintenance time, and the feasibility of standardizing flight control systems. System design requirements and tradeoffs are discussed such as the types of components used, control signal format, method of transmitting signals, actuator configurations, degrees of redundancy, failure detection techniques, and artificial feel mechanization. Examples are given of the application of fly-by-wire control to various classes of aircraft. The primary benefits derived depend on the class of aircraft. Control system technology has reached the point where practical fly-by-wire system designs can be realized today. The next logical step in its development is to build and fly a fly-by-wire system to demonstrate its feasibility and after many flight hours to provide in-flight proof of its maturity.

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14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
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Redundant Systems						
Aircraft Equipment						
Actuators						
Controllers						
Failure Analysis						
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INFORMATION

AD-820427

AFFDL-TR-67-53 July 1967

ERRATA - October 1967

The following correction is applicable to AFFDL-TR-67-53, UNCLASSIFIED Report, July 1967.

Page 15

Delete last paragraph and insert the following paragraph.

Another problem involving control routing and fuel cells occurred in the A-6A. The elevator and rudder control rods in this airplane are routed along the top of the fuselage between the aft fuel cell and the airframe. On one occasion the fuel cell vent stuck while the aircraft was climbing. The tanks over pressurized and the expanded fuel cell then jammed the control linkages against the airframe, thus locking the controls. The pilot was able to place the aircraft on altitude hold, since the longitudinal series servo is integrated into the horizontal stabilizer actuator, and then vent the tank by dropping the landing gear. This freed the controls sufficiently to permit a safe landing. This incident, although one-of-a-kind, illustrates the problem.

Aeronautical Systems Division
Air Force Systems Command
United States Air Force
Wright-Patterson Air Force Base, Ohio